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FEBRUARY 1961

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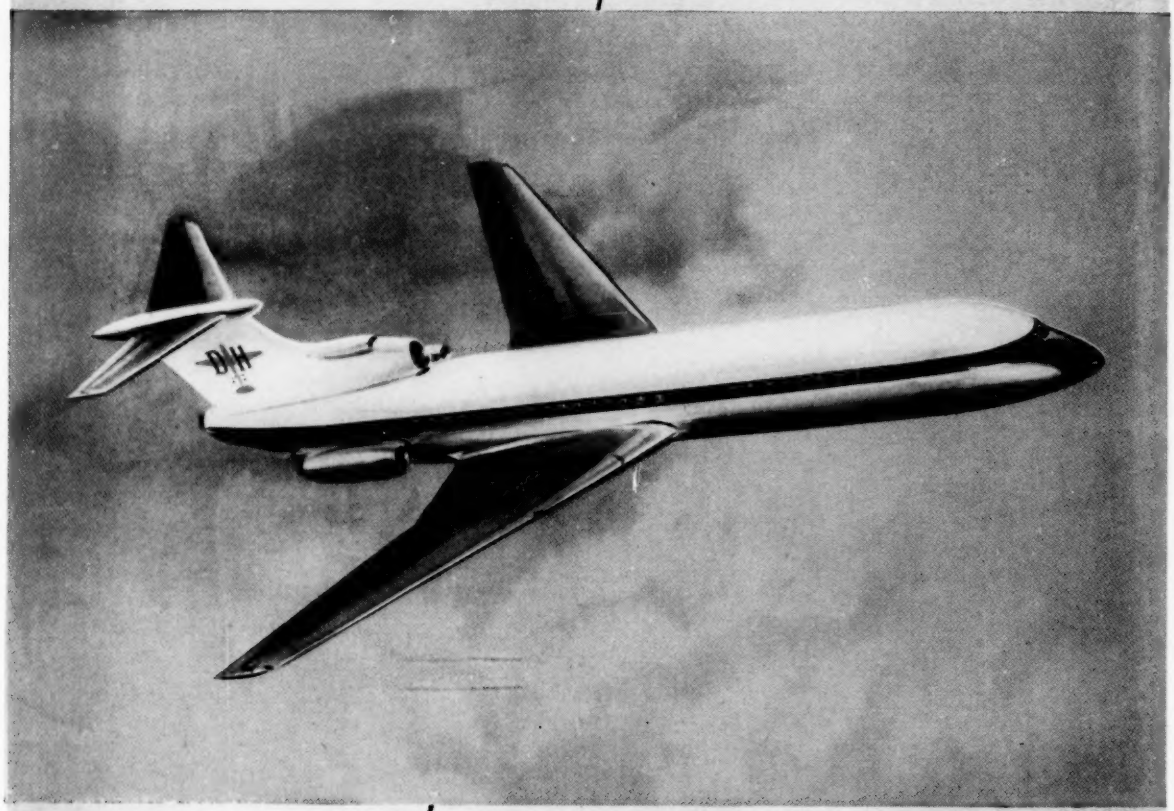

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SUPERSONIC TRANSPORT AIRCRAFT

An All-Day Discussion on Supersonic Transport Aircraft was held by the Society on 8th December 1960 in the Lecture Theatre, 4 Hamilton Place, London, W.1. The Chair was taken by Sir George Gardner, K.B.E., C.B., D.Sc., F.R.Ae.S., M.I.Mech.E., Controller of Aircraft, Ministry of Aviation, and Mr. M. B. Morgan, C.B., M.A., F.R.Ae.S., Deputy Controller of Aircraft (Research and Development), Ministry of Aviation.

Introductory papers were given by M. G. Wilde, B.Sc., A.F.R.Ae.S., Assistant Chief Engineer, Bristol Aircraft Ltd.; D. James, B.Sc., F.R.Ae.S., Chief Structural Engineer, Vickers-Armstrongs (Aircraft) Ltd.; Dr. N. J. L. Megson, Director, Materials Research and Development (Air), Ministry of Aviation; H. Pearson, B.A., F.R.Ae.S., M.I.Mech.E., Chief Engineer (Performance and Research), Rolls-Royce Ltd.; A. V. N. Reed, Deputy Chief Project Engineer, Bristol Siddeley Engines Ltd.; P. L. Sutcliffe, B.Sc., A.F.R.Ae.S., Chief Technician, Advanced Projects Group, Hawker Siddeley Aviation Ltd.; D. M. Jameson, M.B.E., A.F.R.Ae.S., Air Registration Board; Captain F. Ormonroyd, D.F.M., M.A.(Oxon.), F.I.N., Flight Technical Superintendent, Flight Operations Requirements Branch, British European Airways; R. C. Morgan, O.B.E., F.R.Ae.S., Chief Project and Development Engineer, British European Airways; and C. H. Jackson, B.Sc., A.C.G.I., D.I.C., A.F.R.Ae.S., Assistant Chief Engineer (S.D.), British Overseas Airways.

There was a Discussion period after each paper and a General Discussion at the end which was introduced by Dr. A. E. Russell, C.B.E., F.R.Ae.S., F.I.A.S., Director and Chief Engineer, Bristol Aircraft Ltd.

Aerodynamics I

BY

M. G. WILDE, B.Sc., A.F.R.Ae.S.

(Assistant Chief Engineer, Bristol Aircraft Ltd.)

1. Introduction

In a paper delivered before the Society in March 1960, Morgan[1] concluded that a Mach number of about 2·0 may prove a natural "fit" for the first generation of long-range supersonic transports. Subsequently, at the second Congress of I.C.A.S., Kuchemann[2] developed the theme of fitting an aircraft to a particular duty and concluded that the slender wing provides a natural basis for the layout of the supersonic transport. It is convenient to accept these two papers as a starting point for a more detailed examination of the aircraft, since, taken together, they can be taken as fixing some of the major variables—range, speed and general layout.

Kuchemann dealt with the selection of plan form geometry which would yield the maximum lift-drag ratio in the design cruising condition, with certain assumptions concerning the variation of supersonic drag with changes in geometry. This present paper makes these same assumptions but considers also the effect that geometric changes have on the off-design flight conditions.

2. Derivation of Datum Aircraft

Considering the cruise condition, the total drag coefficient may be expressed as

$$C_D = C_{DF} + C_{DW} + C_{DLV} + C_{DLW} \quad . \quad (1)$$

$$= C_{DF} + \frac{512}{\pi}\tau^2 p^2 \left(\frac{s}{l}\right)^2 K_0 + \frac{1}{2\pi} C_L{}^2 \frac{p}{s/l}[K_v + K_w 2\beta^2 (s/l)^2] \quad . \quad . \quad (2)$$

In this expression a convenient set of geometric parameters is used—plan areas, overall length l, overall span $2s$ and total volume of the aircraft V. These are then combined into

s/l	the semi-span to length ratio
$p = \frac{S}{2sl}$	a plan form shape parameter
$\tau = V/S^{3/2}$	a volume parameter.

It will be noted that p is the ratio between the plan area and the circumscribed rectangle and that aspect ratio is not an independent variable since

$$\text{Aspect Ratio } A = 2\frac{s/l}{p}.$$

It is clear from equation (2), which has the normal parabolic form, that the maximum lift-drag ratio will be a function of the geometric parameters s/l, τ and p as well as of C_{DF}, K_0 K_v and K_w. In fact, the plan form parameter p may be taken in combination with K_0, K_v and K_w and values assigned to τ, C_{DF}, S, p^2K_0, pK_v and pK_w which leaves the expression for maximum lift-drag ratio as a function of s/l only for a given Mach number. In one example, Kuchemann has taken

$$\left.\begin{aligned} pK_v = pK_w &= 0{\cdot}7 \\ p^2K_0 &= 0{\cdot}15 \\ C_{DF} &= 0{\cdot}003 \\ S &= 6{,}000 \text{ sq. ft.} \\ \tau &= 0{\cdot}04 \end{aligned}\right\} \quad . \quad . \quad (3)$$

and has then shown that there is a maximum achievable lift-drag ratio at Mach 2·0 of 8·9 occurring at a semi-span to length ratio of 0·202.

This analysis would be taken to indicate that if an aeroplane were being contemplated of such a size that a wing area of 6,000 sq. ft. was appropriate and that the total volume was 18,600 cubic ft., the semi-span to length ratio ought to be chosen as 0·202 in order to generate the maximum lift-drag ratio in cruise. Since this example is to be developed further, the assumptions leading to it must be examined.

In terms of size, the example would represent an aircraft seating about 100 tourist class passengers in a fuselage 165 ft. long. Of the total volume of 18,600 cubic ft., the fuselage would occupy 8,600 cubic ft. and the remaining wing volume would be adequate to hold the fuel needed for a North Atlantic operation (158,000 lb.). The take-off weight would be about 300,000 lb. In these terms such an aircraft is clearly possible.

The drag assumptions are that p^2K_0, pK_w and pK_v are constant. The first of these states that as p is reduced, the zero lift drag increases fairly rapidly. This may be considered reasonable since with p reducing and with fixed fuselage dimensions, the wing fills less of the total plan form area and the drag more nearly approaches that of a body alone. Also the second assumption that the wave drag due to lift increases with

TABLE I

CHARACTERISTICS OF DATUM AIRCRAFT

Geometry		*Weight and Performance*	*Drag*
Plan area	6,000 sq. ft.	Take-off weight 300,000 lb.	$C_F = 0{\cdot}003$
Wing span	66·7 ft.	Zero-fuel weight 142,000 lb.	$K_0 = 0{\cdot}51$
Overall length	165 ft.	Payload 24,000 lb.	$K_v = 1{\cdot}29$
Total volume	18,600 cu. ft.		$K_w = 1{\cdot}29$
		Total fuel 158,000 lb.	
p	0·542	Reserve fuel 30,000 lb.	$(L/D)_m = 8{\cdot}9$
τ	0·04	(1 hr. 40 min. holding)	$(L/D)_c = 8{\cdot}37$
s/l	0·202		
A	0·75		

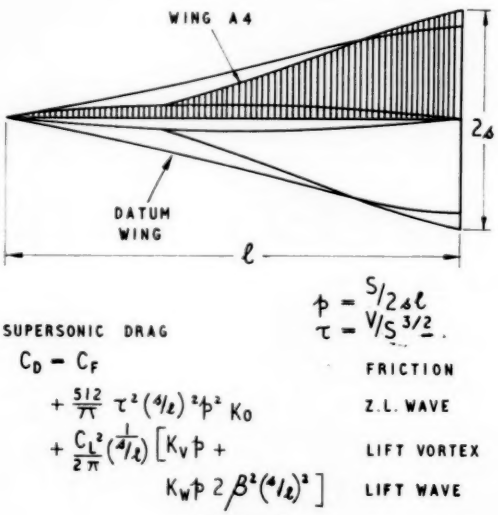

FIGURE 1.

reducing p is reasonable. This reflects the fact that it is difficult to carry load on the small span at the front of a wing of low p. On theoretical grounds, the last assumption that vortex drag should increase with decreasing p is rather more doubtful. However, there is some experimental evidence to substantiate this assumption and it is therefore retained.

In summary then, the optimum aircraft derived in Ref. 2 is reasonable and can be taken to have the characteristics given in Table I.

Of the total fuel carried on this datum aircraft only 51 per cent is burnt in the design cruising condition. The remaining fuel is burnt in the take-off, climb, descent and landing phases of flight and a further amount is carried as a reserve. Of these off-design fuels, there can be argument only about the precise amount to be carried as a reserve. The amount allowed in this analysis is calculated on the basis of 1 hr. 40 min. subsonic holding at altitude and, although this precise manoeuvre may not be required, the fuel quantity is about that which operators believe will be necessary.

3. Variations About the Datum

With a relatively small part of the total fuel burnt in cruise, it becomes unlikely that the datum aircraft above will prove to be the correct choice. It is necessary to establish whether alternative designs offer a better compromise in their overall flight characteristics without change in the duty or in the economic result. Improvements may be found in several areas—speed and controllability on the approach, airfield performance and fly-over noise for example—and, provided such improvements are achieved at no significant economic penalty, the compromise aircraft is more attractive.

In studying variations about the datum, the fuselage geometry, overall volume and take-off weight have been maintained constant. The first condition fixes the payload. If the fuel calculated for the stage does not vary greatly, then constant volume ensures adequate

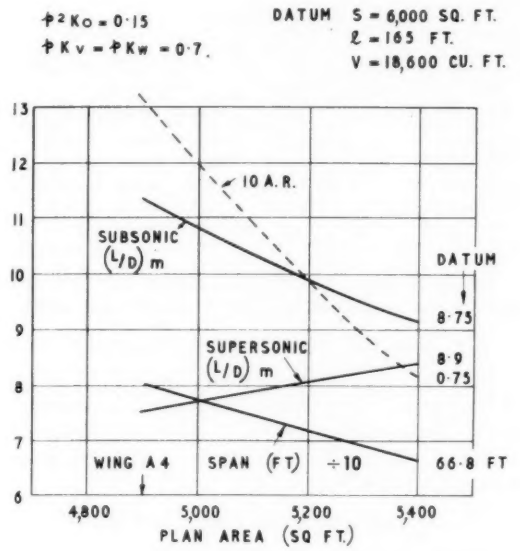

FIGURE 2.

stowage space for fuel in the wing. Finally, the direct operating costs at fixed payload vary directly with take-off weight to a first approximation so that holding this constant ensures similar costs for each variant.

TABLE II

CHARACTERISTICS OF AIRCRAFT VARIANTS

T.O.W.	= 300,000 lb.	Range=Trans-Atlantic
Overall volume	= 18,600 cubic ft.	Reserves: 1 hr. 40 min. subsonic holding
Overall length	= 165 feet	$C_{DF}=0{\cdot}003$,
Payload	= 24,000 lb.	$pK_v=pK_w=0{\cdot}7$
		$p^2K_0=0{\cdot}15$

	Datum	*Variations from Datum*			
	A	*A*1	*A*2	*A*3	*A*4
(*a*) Aircraft					
S (sq. ft.)	6,000	5,400	5,250	5,100	4,900
s ft.	33·4	33·4	35	37·5	40
(s/l)	0·202	0·202	0·212	0·227	0·242
p	0·542	0·490	0·455	0·412	0·371
τ	0·04	0·0467	0·049	0·0512	0·0543
A	0·75	0·825	0·933	1·100	1·305
(*b*) Supersonic cruise					
K_0	0·51	0·625	0·725	0·890	1·085
K (total)	1·605	1·777	1·947	2·217	2·540
$(L/D)_m$	8·90	8·38	8·19	7·85	7·48
$(L/D)_c$	8·37	7·87	7·70	7·38	7·03
Engine scale	1·0	1·018	1·073	1·169	1·280
(*c*) Subsonic flight					
D_0/q	33·05	29·86	29·44	28·75	27·81
$(L/D)_{max.}$	8·75	9·18	9·65	10·43	11·35
(*d*) Weights (in lb.)					
Zero fuel	142,000	138,500	139,500	142,000	144,500
Off-design fuel	76,000	74,500	70,500	65,500	56,500
Cruise fuel	82,000	87,000	90,000	93,500	99,000
Take-off weight	300,000	300,000	300,000	300,000	300,000
Landing weight	172,000	166,500	166,000	167,000	167,500

Four variations have been examined within these given boundary conditions by progressively decreasing the wing area and increasing the aspect ratio from the datum case. In terms of non-dimensional parameters, (s/l) is increased, p is decreased and τ is increased. The effect of these changes can be described in general terms. For progressive decrease in p,

(*i*) the maximum lift-drag ratio at $M=2{\cdot}0$ decreases. The increase in τ at fixed span to length ratio has this effect and, additionally, an increase in (s/l) above the optimum reduces $(L/D)_m$ still further. It is assumed that the aircraft is cruised at a fixed percentage of $(L/D)_m$ so that this leads directly to an increase in cruise fuel,

(*ii*) the power plant weight increases since it is assumed that the cruise condition fixes the engine size,

(*iii*) the structure weight reduces through reduction in wing area,

(*iv*) the off-design fuels decrease through reductions in subsonic $k/\pi A$ with increasing aspect ratio.

The first two of these items would lead to an increase in the take-off weight and the second two items offset these effects. By a judicious choice of the geometric parameters, it is possible to obtain variations about the basic design for which the take-off weight is unchanged.

Table II summarises the characteristics of the four variations studied.

4. Comparison of Flight Characteristics

All the variants above perform the given duty at the same take-off weight. The take-off and landing wing loadings vary, however, and so does the aspect ratio. These parameters affect the flight characteristics in terms of speed and control power on the approach, airfield performance and fly-over noise as indicated below.

4.1. APPROACH

None of the wings considered here will stall in the accepted sense so that approach speed will be limited by other considerations. One consideration will certainly be that of attitude as it affects pilot visibility and passenger comfort. It is proposed to define here a " minimum flying speed " at an incidence of 18° above the zero lift condition in free air and to take the approach speed as being 25 per cent above this minimum flying speed. This is equivalent to an attitude of 12° above no-lift on the approach.

Ref. 2 gives an equation for the low speed lift curve accounting for the non-linear lift contribution.

$$C_L=\frac{\pi}{2}A\alpha+2\alpha^2 \qquad . \quad . \quad (4)$$

Using equation (4) and the landing weights given in Table II, the minimum flying speeds and approach speeds are as given in Table III.

Table III shows a clear advantage for the aircraft of higher aspect ratio in terms of approach speed. These compromise designs become more attractive when the controllability in cross winds is examined.

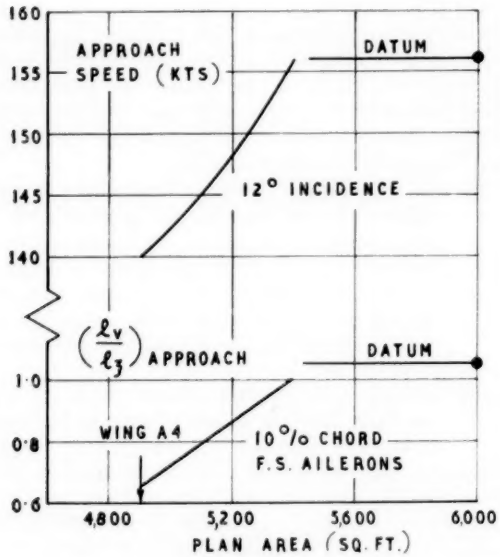

FIGURE 3.

Ref. 3 gives some measured coefficients of rolling moment due to sideslip (l_v) and rolling moment due to trailing edge aileron control (l_ξ) for a series of delta wings of varying aspect ratio. These results have been corrected to a common basis—control chord 10 per cent of root chord extending over the full span. The ratio of (l_v/l_ξ) is used as a measure of the ability to control in a cross-wind landing in Table IV.

These results are themselves only comparative but, when interpreted for realistic designs where full span differential aileron control is not available, it may well be the case that the datum aircraft is unacceptable whereas aircraft *A*3, for example, would pass.

4.2. AIRFIELD PERFORMANCE

In arriving at the weights and performance given in Table II, it was assumed that sufficient engine thrust was installed to fly each aircraft at 94 per cent of its maximum cruising lift-drag ratio. Given a type of engine, turbo-jet or turbo-fan for example, it is then possible to derive the total sea-level static thrust installed in the aircraft and to use this quantity in evaluating the take-off field length required.

TABLE III

APPROACH SPEEDS

Aircraft	*Datum* *A*	*A*1	*A*2	*A*3	*A*4
Min. flying speed (kts.)	125	125	121	116	112
Approach speed (kts.)	156	156	151	145	140

TABLE IV

CONTROL IN CROSS WINDS

Aircraft	*Datum* *A*	*A*1	*A*2	*A*3	*A*4
(l_v/l_ξ)	1·05	1·00	0·90	0·80	0·65

TABLE V

TAKE-OFF DISTANCE

Aircraft	*Basic* *A*	*A*1	*A*2	*A*3	*A*4
$C_L u/s$	0·374	0·402	0·444	0·508	0·586
T_s/1,000 (lb.)	97·5	99·0	104·5	114·0	124·5
s_{35} (ft.)	9,250	9,420	8,300	6,850	5,670
T.O. distance (ft.)	11,100	11,300	9,960	8,220	6,800

In this paper, the following engine law is assumed, appropriate to an unreheated turbo-jet engine.

$$\text{Static Thrust } (T_s) = 2{,}250\,(T/q) \text{ cruise} \quad . \quad (5)$$

It is also necessary to estimate the unstick speed, or lift coefficient. The unstick lift coefficient is taken here as that appropriate to an incidence of 10° from no lift in ground effect. Using the form of lift curve given by equation (4) this becomes

$$C_L u/s = 1{\cdot}4\,(0{\cdot}275A + 0{\cdot}061) \quad . \quad (6)$$

Finally, the all-engine distance to screen height may be expressed in terms of total installed static thrust T_s, take-off weight W and unstick lift coefficient in the approximate form

$$s_{35} = 225W^2/ST_sC_Lu/s \quad . \quad . \quad (7)$$

Using equations (5), (6) and (7) and the known thrust requirements in cruise, the all-engine distances to 35 ft. have been calculated for each of the variants in Table V. Take-off distances required are also given on the assumption that these will exceed s_{35} by 20 per cent.

The higher aspect ratio wings show a considerable advantage in take-off distance. The order of merit is again the same on landing distance required but the landing performance is very non-critical on any aircraft in this class provided that effective thrust reverse is made available.

4.3. FLY-OVER NOISE

It may be supposed that a normal type take-off flight path is followed from the screen height consisting of an initial full throttle climb at unstick speed followed by a throttled climb out from above the observation point. Here the observation point is taken to be 4·2 statute miles from start of roll. The gross gradient of climb is 3 per cent after throttling and the thrust required is T_R. The engine noise data used are typical of those for a straight turbo-jet engine of relatively high specific thrust.

These absolute values of noise can be improved in

TABLE VI

FLY-OVER NOISE

Aircraft	*Basic* *A*	*A*1	*A*2	*A*3	*A*4
Height at 4·2 miles (ft.)	630	635	1,150	1,830	2,635
Thrust T_R/1,000 (lb.)	84·0	84·0	80·0	78·2	74·8
Fly-over noise (PNdB)	132½	132½	123½	117	111½

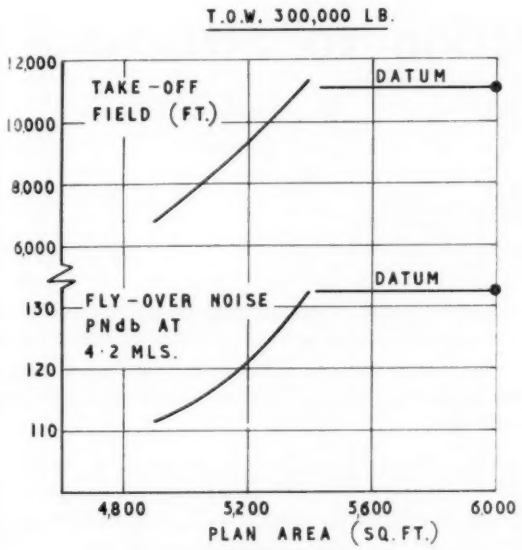

FIGURE 4.

many ways, but the very significant differences between the datum aircraft and those of higher aspect ratio will remain.

In almost all of the off-design flight characteristics considered, the datum aircraft, chosen for its high cruising efficiency, is considered to be unacceptable. The analysis shows, however, that shapes are available which show significant improvements on the datum and it would be from among these that the designer would look for the final compromise. It seems inevitable that the aerodynamicist must abandon his ideal configuration and face the problems of supersonic flow past shapes where the wing and fuselage are distinguishable features of the layout.

5. Interference Fields

In the example aircraft *A*3, the gross plan area was 5,100 sq. ft. and the total volume 18,600 cubic ft. If it is assumed that the passengers are contained at 4-abreast in a circular fuselage of high fineness ratio, about 10 ft. in diameter, the plan area of the fuselage would be approximately 1,000 sq. ft. and its volume 8,600 cubic ft. Therefore the wing clear of body occupies an area of 4,100 sq. ft. If this wing had a constant thickness-chord ratio over the whole span, its maximum depth would be about $5\frac{1}{2}$ ft. at the body side, *i.e.* about 60 per cent of the body depth itself. It is clear, therefore, that the fuselage flow field will significantly affect the wing flow field, particularly at supersonic speeds.

It is well-known that the interference field arising from bodies attached to a lifting wing can be made to be favourable in terms of lift-drag ratio. By assuming that the effect of an interference field is to displace the origin of the normal drag polar, it is possible to examine the possibilities in this area. For aircraft *A*3, the drag polar at cruise speed was assumed to be

$$C_D = 0{\cdot}00633 + 0{\cdot}641 C_L^2 \quad . \quad . \quad (8)$$

TABLE VII

MAXIMUM LIFT-DRAG RATIOS WITH INTERFERENCE

C_{L_0}	0	0·0025	0·005	0·0075	0·01
$(L/D)_m$	7·85	7·97	8·09	8·21	8·33

If the origin were moved up by an amount ΔC_{D_0} and to a value of C_L denoted by C_{L_0} then the equation becomes

$$C_D = 0{\cdot}00633 + \Delta C_{D_0} + 0{\cdot}641 (C_L - C_{L_0})^2 \quad (9)$$

It can then be shown that the maximum lift-drag ratio is given by

$$(L/D)_m = 7{\cdot}85\left(1 - \frac{\Delta C_D}{0{\cdot}01266} + \frac{C_{L_0}}{0{\cdot}0993}\right) \quad (10)$$

where 7·85 is the maximum lift-drag ratio with the original drag polar occurring at a lift coefficient of 0·0993.

A successful application of these principles might yield a result in which the drag increment ΔC_D was about one-twentieth of the value of C_{L_0} and, in this case, the lift-drag ratios of aircraft *A*3 would be those given in Table VII.

It is clear from these values that there is a very substantial increase in cruising efficiency to be obtained by arranging that the wing and fuselage flow fields interfere in a favourable way.

6. Conclusions

By assuming a certain form of variation of zero-lift wave drag and lift dependent drag with changes in geometry, it is possible to derive the proportions of a supersonic transport aircraft having the maximum possible cruising lift-drag ratio. This datum aircraft would have its lift distributed over the same length as its volume and the subsonic and supersonic flow fields would be basically simple in form.

This present paper shows, however, that this simple basic design will have poor low-speed flight characteristics. It is also shown that other layouts are available which achieve the same economic results and have greatly improved off-design flight characteristics and it will be from these that the designer will choose the final aircraft.

These compromise layouts have reduced wing area and increased aspect ratio as compared with the datum aircraft and the wing and fuselage then become distinguishable features of the design. It follows that the aerodynamicist will be forced to consider more complex flow conditions in which there are substantial interference effects present. One of the prime tasks will be to use these interference fields to the best possible advantage in the cruise condition.

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Kinetic Heating in Relation to Structural Integrity

BY

DAVID JAMES, B.Sc., F.R.Ae.S.

(Chief Structural Engineer, Vickers-Armstrongs (Aircraft) Ltd.)

1. Introduction

This short paper is confined to a discussion on one aspect of the Structural Design of Supersonic Transports. It is chosen because it covers the area in which the impact of supersonics is probably most critical from the structural point of view, *i.e.* kinetic heating.

It is of utmost importance that the significance of kinetic heating is not exaggerated. Adequate attention must be paid to assessing the degree to which past experience is relevant if excess weight is to be avoided. It is in this direction that the discussion is aimed.

In view of the need to keep the paper short, attention is concentrated on the Mach 2 class of aeroplane constructed largely of aluminium alloy. Much of what is said will also have relevance to the Mach 3 class of aeroplane.

2. The Basic Structural Integrity Requirements

The requirements for the structural integrity of civil aircraft are specified in terms of proof and ultimate load carrying capabilities. The primary structure is required to be capable of:—

(*i*) supporting all loads up to and including the proof load so that such elastic and permanent deformation as occurs during application of the load does not interfere with the safe operation of the aeroplane;

(*ii*) supporting the ultimate load.

There is also a fatigue requirement that the strength and fabrication of the aeroplane throughout its operational life shall be such as to ensure that the possibility of disastrous fatigue failure of the primary structure is exceedingly remote under the action of the repeated loads of variable magnitude expected in service.

The designer has to satisfy himself and the airworthiness authorities that these basic requirements are fulfilled in the new conditions.

3. The Effects of Kinetic Heating

Kinetic heating brings in its train three factors which have hitherto not been of great concern, namely:—

(*i*) The effect of high temperatures on the mechanical properties of the structural materials.

(*ii*) The development of a self-equilibrating load system due to temperature gradients through the structure.

(*iii*) The added effect of creep under sustained load on the permanent deformation of the structure.

These three aspects will now be considered in turn, with emphasis on the degree to which past experience is relevant.

3.1. THE EFFECT OF TEMPERATURE ON STATIC STRENGTH

The ultimate strength requirement includes a safety factor of 1·5 to cover a range of uncertainties, including the accuracy of estimation of loads, stress distribution and failing stress. In the case of the supersonic transport aircraft accuracy of estimation of temperature must be added to the list. Does this require an increase in the safety factor? Only if the errors of temperature estimation are of greater significance than past experience shows to be adequately covered by the existing factors.

Figure 1 is a familiar presentation of the effect of temperature variation on material properties which will not be discussed further.

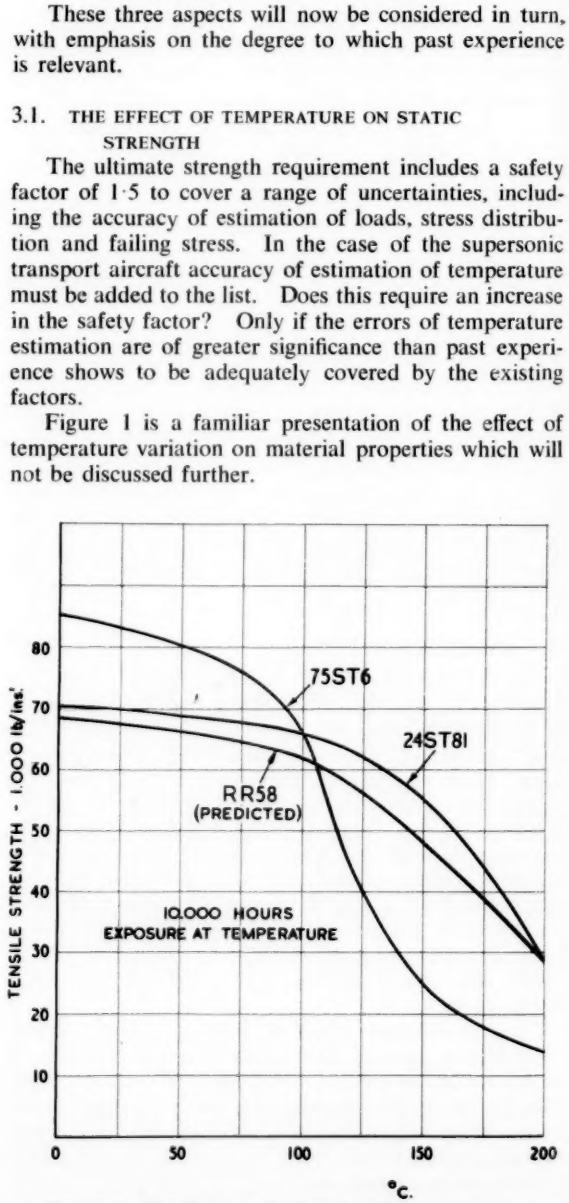

FIGURE 1. Tensile strength plotted against temperature.

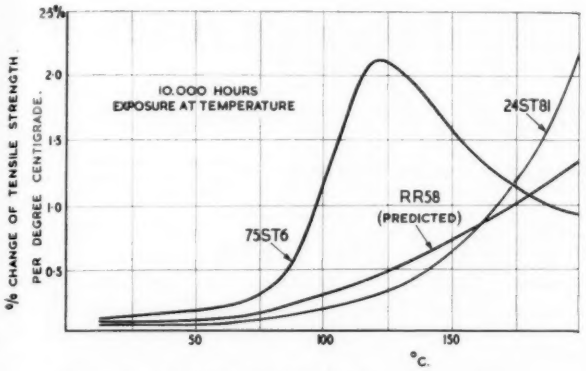

FIGURE 2. Tensile strength variation at temperature.

Figure 2 is derived from Fig. 1 to present the rate of change of material properties with temperature at a given temperature. The ordinate shows the percentage error in strength estimation per degree error in temperature estimation.

The data for three well-known structural materials are plotted. It will be noted that the curve for the aluminium zinc alloy shows 0·1 per cent per °C variation at 25°C increasing to 0·3 per cent per °C at 75°C and then rising rapidly to a peak of rather more than 2·0 per cent at about 140°C. This is an alloy which has been extensively used for structural purposes on subsonic aircraft and strengths corresponding to the 25°C values have been used for design purposes. In practice, temperatures 15°C higher than this are experienced quite frequently and in the tropics structural temperatures of up to 75°C have been recorded. These temperatures would result in tensile strength reductions of the order of 1·5 and 5 per cent respectively. It has been accepted that these reductions are adequately catered for by the overall safety factors currently used and as far as is known the airworthiness record does not suggest that this has been an unsound procedure.

Comparing now the curves for the other alloys 24S and RR.58 with that of the zinc alloy, it will be noted that the percentage error per °C increases much more slowly, the figure of 0·3 per cent per °C not being reached with the 24S alloy until the temperature is raised to 120°C. The structural temperature likely to be reached on an aircraft cruising at Mach 2·2 is of this order. Higher temperatures than this will only be reached on the occasional overshoot. An examination of the data corresponding to 100 hours exposure shows that at 160°C the error is only 0·5 per cent per degree.

The supersonic air liner will almost certainly fly to a temperature placard limitation. It is suggested, therefore, that no case exists for the inclusion of a special safety factor to cover the effects of errors of temperature on ultimate material strength provided the right type of alloy is chosen.

Similar curves showing the same trends can also be produced for the material proof strength and modulus of elasticity.

TABLE I

Stress level	*Modulus of elasticity*
Up to L.P.	1·0 *E*
At 0·1 per cent proof stress	0·3 *E*
At 0·2 per cent proof stress	0·2 *E*
At 0·5 per cent proof stress	0·1 *E*

3.2. THE EFFECT OF TEMPERATURE GRADIENTS

The effect of a temperature gradient in the structure is to induce a self-equilibrating stress system. This is necessary to achieve compatibility of strain. The magnitude of the stresses involved depends upon relative areas, relative moduli of elasticity and temperature difference. The temperature difference will depend on thermal conductivity, radiation characteristics and so on.

From the point of view of the present discussion, attention will be confined to consideration of the two questions:—

What is the significance of the temperature gradient from the point of view of the basic airworthiness requirements and how relevant is past experience in the assessment of the need for special safety factors to cover uncertainties of temperature estimation?

Thermal stresses are induced in achieving compatibility of strain and their magnitude decreases as the modulus of elasticity decreases. The shape of the stress-strain curves for the aluminium alloys is well known. An examination of the curves will show that the modulus of elasticity varies with the stress approximately as shown in Table I.

It will be obvious from these values that a given temperature gradient will become rapidly less significant from the point of view of the stress induced once the total stress due to external load and temperature gradient exceeds the limit of proportionality. As the ultimate strength of the material is approached the thermal stresses become insignificant.

The ultimate failure of a component may arise from one of two basic causes, a metallurgical failure of the material or elastic instability. Elastic instability will nearly always occur at stress levels at which the modulus of elasticity is equal to Young's modulus and it is obvious that in such cases the presence of a temperature gradient will significantly affect the external load the component can carry.

It is suggested, therefore, that a factor of safety should be applied to the thermal strain. If this is taken as 1·5 then where failure is within the elastic range the stresses induced are treated in exactly the same manner as the stresses due to external loads. This is also compatible with the proof strength requirement, which is in reality a proof strain requirement.

A further factor to be resolved is whether thermal stresses should be assessed on the basis of the estimated temperature gradient or whether a factored temperature gradient should be used.

Current civil aircraft types are operating with structural temperature differences probably of the order of 75°C and occasionally perhaps up to 90°C. As

stated earlier, structural temperatures of 75°C are achievable on aircraft surfaces exposed to the tropical sun.

Fuel may well be loaded at a temperature of at least 40°C. Statistics supplied by the British Petroleum Company show that this order of temperature is the mean value reached in ground storage tanks in some areas for as much as three months of the year. A subsonic jet transport carrying fuel loaded at this temperature is likely to have wing skin temperatures at the end of the climb of about −35°C, allowing 25°C for kinetic rise above ambient. There is thus a temperature difference of 75°C under such conditions.

Similarly, on a long flight the structure may be soaked at, say, −35°C and at the end of the flight the aircraft may descend to conditions where temperatures of 40°C will be reached on the exposed skins.

Temperature stresses due to these effects have not normally been regarded as significant.

It would be reasonable, therefore, to accept that our airworthiness standards will not be reduced if thermal stresses are assessed on the best available data without recourse to applying additional factors on temperature gradients.

The discussion so far has been solely concerned with the static strength implications of thermal stresses. It is likely that the fatigue implications will be far more critical. The reasons for this are twofold. Firstly, the stresses induced may well be of the same order as the l_g wing stresses and l_p fuselage hoop stresses experienced on current aircraft which have already been shown to be major contributors to fatigue damage. Secondly, the most critical areas are likely to be hidden away inside the structure and early detection of failures thereby rendered more difficult.

It is particularly important, then, that adequate attention is paid to the fatigue implications of thermal stresses.

3.3. CREEP

Creep has long been a factor of considerable significance to the engine designer and alloys have been developed in which creep resistance has been the major aim. It is, however, a new problem to the structural designer and the background provided by the engine designer's experience is limited to what must be regarded as short time creep in the airframe context.

Superficially it would appear that provided the steady stress level during the supersonic flight phase is kept below some critical value, depending upon the temperature and the material used, there will be no problem. Some arbitrary limit as to the magnitude of the permissible creep needs to be defined in order to assess the critical stress level.

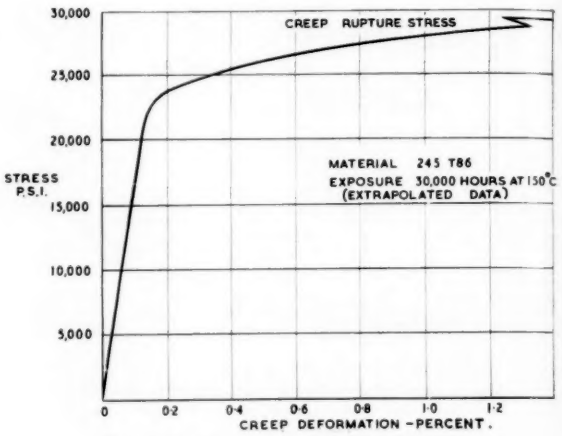

FIGURE 3. Stress-creep deformation curve.

Figure 3 shows the probable stress-creep deformation curve resulting from 30,000 hours exposure of 24.ST.86 alloy at 150°C.

It must be emphasised that the curve illustrated is based on extensively extrapolated data and it is only used to indicate trends. It has the shape of a stress-strain curve. The curve indicates that if a reasonable margin of safety against creep rupture is to be obtained, then a permanent creep deformation of about 0·1 to 0·2 per cent is all that can be permitted to develop during the life of the aeroplane.

It is also apparent that such a limitation is unlikely to be an embarrassment as the sustained stresses will be limited for other reasons to lower values than those which would produce these orders of creep deformation.

There is, however, another problem. All parts of the structure will not be under the same stress or at the same temperature, and creep deformation will, therefore, develop at different rates. The result will be the progressive build-up of a self-equilibrating stress system which may well be significant. Buckling characteristics of skin panels may change and the pattern of fatigue stresses be altered by a continuous change in local mean stress levels. These aspects of creep are the ones about which least is known and on which the greatest attention should be concentrated.

ACKNOWLEDGMENT

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The views expressed are the personal opinions of the author and do not necessarily reflect the policies of the Company.

Materials—Metallic and Non-Metallic

BY

DR. N. J. L. MEGSON

(Director, Materials Research and Development (Air), Ministry of Aviation)

1. Introduction

With increasing speeds of aircraft, problems arise over materials of construction, partly through kinetic heating effects which cause temperature rises and partly through rain erosion effects. Additionally, account has to be taken of a required life of 30,000 hours which is much longer than that over which most materials are normally evaluated.

For speeds of Mach 2·2, the maximum temperature attained will be of the order of 130°C, while for Mach 2·7 the corresponding temperature will be 200°C or higher. For the lower speed, it is probable that aluminium alloys will be satisfactory, although it must be emphasised that no information is yet available on the long-term behaviour of materials at the elevated temperatures likely to be experienced. Test programmes are in hand, but these will necessarily take considerable time for their completion. For speeds of Mach 2·7 or above, steel or titanium alloys will be required, but here again the necessary evaluation for long periods at high temperature has yet to be completed.

The non-metallic field poses different problems. Up to temperatures of 130°C, it is likely that existing rubbers and plastics for tyres, seals and radomes will be adequate, although difficulties will appear in the case of transparencies. At higher temperatures, non-metallics will generally give trouble; existing commercially obtainable rubbers are unlikely to be usable above about 150°C, glass-reinforced radome materials will be of marginal value at 200°C, and the transparency problem will be accentuated. Furthermore, protection of radomes against rain erosion becomes increasingly difficult as speed and temperature increase.

Following these general remarks, it is proposed to consider metals and non-metals in more detail, indicating their known properties, their potential usefulness, and means by which they may be improved or modified for high temperature use.

2. Metallic Materials

2.1. ALUMINIUM ALLOYS

Three aluminium alloys in quantity production are in common use for subsonic aircraft. Two of these, namely L.73 (containing copper and small amounts of manganese and magnesium) and D.T.D.687 (containing zinc and magnesium and small amounts of manganese and copper) are British. The third, Clad 2024, is American and contains copper with minor amounts of magnesium, manganese, silicon and iron. D.T.D.687 is the strongest at ordinary temperatures, having a tensile strength of 32 tons/in.² with a 0·1 per cent proof stress of 27 tons/in.² L.73 has a tensile strength of 27 tons/in.², while 2024 is intermediate. D.T.D.687 needs care in use because of its susceptibility to stress cracking and fatigue failure.

The static properties at room temperature unfortunately give no guide to those which may be observed after long time soaking at elevated temperatures. Thus D.T.D.687 loses its superior proof stress after 100 hours soak at 150°C and it shows the lowest recovery proof stress at that temperature after 80 hours soak.

No creep data are available for D.T.D.687. It is found, however, that the creep resistance of 2024 in the T4 condition is better than that of L.73 for periods of between 100 and 1,000 hours at temperatures of 130°, 150° and 200°C.

Three copper-bearing alloys, namely RR.57, RR.58 and X2020 (an American product) have been developed for service at elevated temperatures. RR.57 (which includes minor amounts of manganese and titanium) was devised as a forging alloy for pistons and compressor blades with service temperatures as high as 350°C; its production in the form of clad sheet depends on demand. It has a tensile strength of 23 tons/in.², a 0·1 per cent proof stress of 15 tons and an elongation of 10 per cent. RR.58 (containing magnesium, nickel, iron and titanium) was also developed as a forging alloy, in this case for engine components functioning at 300°C, although small amounts of clad sheet are being produced for evaluation. Its tensile strength is 25 tons, with a 0·1 per cent proof stress of 20 tons and an elongation of 6 per cent. X2020 is unusual in containing 1·1 per cent of lithium in addition to manganese and cadmium. It shows the tensile strength properties of the zinc-magnesium bearing alloy at room temperature combined with those of the copper-bearing alloys at temperatures up to 150°C for short periods. Its modulus of elasticity is about 6 per cent higher than that of normal high strength aluminium alloys and its density is 2 per cent lower. It can be obtained in the form of bars, forgings, extrusions and sheet. Its fatigue strength is barely comparable with that of other alloys and unfortunately it is notch sensitive under tensile stress.

Efforts are being made to evaluate certain of these alloys more precisely for long time service at elevated temperatures. Tensile and compression tests at various temperatures after soaking at elevated temperatures for 30,000 hours, tensile creep tests up to 150°C for 20,000 hours, crack propagation tests up to 150°C and tensile tests on riveted joints up to 150°C are expected to provide much of the necessary design data. Because many of these determinations will continue for 3½ years before full results are obtainable, attention is being directed to the possibilities offered by extrapolation formulae of the Larson-Miller type. These may provide a means

of predicting long time behaviour of alloys at elevated temperatures from room temperature properties. The Royal Aircraft Establishment has been active in this field and has developed refinements of the Larson-Miller equations.

With regard to the possible improvement of high temperature tensile strength combined with satisfactory creep resistance the metallurgical structural requirements are similar. It is most desirable to provide stable coherent precipitates and to arrange for precipitation of a strengthening phase at the operating temperature and for slow diffusion rates in the matrix. Grain size control by the introduction of finely dispersed oxides as in S.A.P. material gives certain high temperature improvements, although the room temperature properties are unsatisfactory. Improvements might be brought about through the introduction to such structures of elements which would reduce the stacking fault energy of the aluminium matrix.

2.2 STEELS

For the supersonic aircraft, the steel under chief consideration is the Firth Vickers 520 (Fe-16 Cr-6 Ni-1·5 Cu-1·5 Mo-0·3 Ti-0·07C). This is a precipitation-hardening stainless steel, in which titanium serves as a stabiliser against weld decay. Tensile strengths vary between 65 and 85 tons/in.^{2} according to thermal and rolling treatments. It is available as 40 in. wide strip in the thickness range of 0·08 to 0·02 in. Further development is needed on the production of 0·01 in. strip, 40 in. wide and of acceptable flatness.

For evaluation, sheet will be used which has been processed by numerous variants of transformation temperatures, cold stretching and hardening temperatures to determine the best conditions for producing wing and fuselage skins, straight stiffeners and pressed frames. Typical properties for evaluation include stress-corrosion, crack propagation, tensile and compressive strength and limited creep tests, supplemented by fatigue, crack propagation, transverse shear strength and weld decay tests on spot-welded material.

For aircraft usage, improvements in the elastic moduli (both E and G) are desirable, insofar as they would help in stiffening the structure and in increasing the strength of compression panels. A reduction in density ρ would also be valuable, because material efficiency in lightly-loaded compression structures is proportional to E/ρ^3. The densities of the common alloying constituents are, however, so high that significant reductions in ρ are unlikely except by drastic changes in composition. A possibly hopeful line is the development during the past five years of alloys containing up to 10 per cent of aluminium. The British alloy has a density of 6·75 gm./c.c. Unfortunately such alloys corrode rather rapidly, although it is possible that corrosion resistance may be significantly improved by further development.

2 3. TITANIUM

Titanium alloys could bridge the gap between Mach 2·2 for aluminium alloys and 2·7 for steel. The best titanium alloys possess greater specific strength than stainless steels in the temperature range 0-300°C, better efficiency in lightly loaded compression structures and superior corrosion resistance. Against these advantages, however, must be offset the considerably higher cost (20 times that of FV.520), slightly lower specific stiffness, a more restricted range of sheet sizes and thicknesses, and production and fabrication techniques less well developed.

The only available high strength sheet alloy in this country is a Ti-15 per cent Mo, which is heat treatable to a strength of 70-80 tons/in.^{2} Its density (4·95 gm./c.c.) is about 10 per cent greater than an American alloy (Ti-4 A1-3 Mo-1V). The strength of the British alloy is 15 per cent lower than the American alloy up to temperatures of 300°C. The production and fabrication techniques of the American alloy leave much to be achieved in terms of size, flatness and surface finish.

Full use needs to be made of the potentiality of the current British alloy, but as a general conclusion, considerable development is required on new high strength titanium alloys (including possibly the Ti-6 A1-4V now available as a medium strength alloy), and also on production and fabrication techniques.

An increase in modulus and a reduction of density are desirable. Investigations at the Royal Aircraft Establishment have shown that a Ti-5 A1-1·5 B alloy has a 26 per cent superiority in specific modulus over the stiffest commercial titanium alloy and a smaller, but significant, advantage over the stiffest stainless steel.

3. Non-Metallic Materials

Although non-metallic materials are used in aircraft to a minor extent as compared with metals, their adequate performance is vital to the efficiency and safety of the aircraft. No particular difficulties arise for speeds up to about Mach 1·0, but above that speed problems associated with stability and strength begin to appear. Transparencies, radomes and applications involving rubber-like materials (elastomers) all have their special problems.

Most of the non-metallic materials in use are organic in character. Thermal breakdown may occur at comparatively low temperatures either through softening or decomposition; both mechanisms lead to severe loss of strength. Efforts are being made to improve heat-resistance by the synthesis of new organic products, but it may be necessary to look to the almost unexplored fields of inorganic or semi-organic polymer chemistry for worthwhile advances.

3.1. TRANSPARENCIES

The present organic material, of Perspex type, is not usable above about 90°C, because of softening, so alternatives must be sought. Certain lightly cross-linked polymers have been suggested, the chief representative in this country being Avialite. It is more brittle than Perspex and its engineering may well be difficult. It is under examination, although experience with it is limited. It could be suitable for temperatures of 130°C, but is unlikely to be usable at 200°C.

Inorganic glasses or fused quartz may have to be

employed, with the attendant difficulties associated with double curvature and thermal shock behaviour still to be solved. Results from the prolonged loading of glass at room temperature indicate a possible usage up to 10,000 hours, but similar data at 100°C or above are lacking. At 200°C some stress relaxation in existing toughened glass would be expected over extended periods of time.

A sandwich construction, with a rubbery interlayer, constitutes another line of attack, but few data on such products are available.

3.2. RADOMES

Existing materials based on glass-fibre-reinforced resins should prove adequate for temperatures up to 130°C, but no estimate of probable life is yet possible. At 200°C or above, ceramics will probably be required; fabrication to fine tolerances and thermal shock behaviour constitute the chief problems. Rain erosion difficulties may be overcome by suitable radome design or by the adoption of an appropriate flight plan.

3.3. ELASTOMERS

Elastomers provide the most difficult problems among non-metallic materials. Little information exists on the long-term behaviour required and in many cases a high replacement rate must be expected. Chemical changes, accompanied by stress relaxation, will occur in many applications, causing loss of interference of seals, nose end-fittings and so on.

Conventional natural or synthetic rubbers have very limited use at 100-130°C. The best of them—Butyl rubber—softens appreciably in 14 days at 150°C and loses 50 per cent of its strength. Others embrittle rapidly.

There are three special elastomers now available, but not necessarily in this country. Silicone rubbers show good performance in air at 205°C for more than 1,000 hours. Their use in fluids is limited, and they have relatively poor compression set properties, but they may be used with glass or asbestos in gaskets to limit creep. A fluorinated silicone, LS.53 has improved resistance to certain fluids, but is inferior to the unmodified silicones at high temperature. Viton A is a new fluorinated elastomer which, according to American reports, remains flexible after 240 hours at 250°C or 30 hours at 300°C and is useful in ester lubricants after 1,000 hours at 150°C. Experience of this American product is limited.

The plastic P.T.F.E. (Fluon, Teflon) has high thermal stability combined with limited rubberiness, although cold flow is observed at 150°C.

Tyres

At temperatures between 100° and 130°C Butyl rubber reinforced with steel wire may be adequate for a limited number of landings, though increased landing speeds will aggravate the difficulties. Refrigeration is desirable and will be essential at high speeds, if conventional undercarriages are used.

Fuel system

Integral tank sealants may constitute a severe problem, especially if the fuel is used as a heat sink. Accessibility for maintenance is inevitably poor. A polysulphide rubber sealant operating at 150°C is known, but its life is doubtful.

Cabin sealants

Considerations similar to those for fuel systems apply unless the pressure cabin wall is inside the refrigerated zone.

Seals etc.

Silicone rubbers will be useful for dry seals; the choice of fluid seal materials will depend on the fluid or lubricant selected.

Hydraulic hose

Hose are subject to considerable flexure on the undercarriage system and embrittlement is a hazard. They would benefit from refrigeration: type of rubber selected again depends on fluid.

4. Conclusions

It seems likely that aluminium alloys will be used for a Mach 2·2 aircraft, and steel or titanium alloys in the Mach 2·7 case. Extensive test programmes are proceeding with the idea of obtaining design data. For future use, it would be valuable to develop a parameter for the prediction of long-term behaviour at elevated temperature from short-term tests. Metallic materials problems arising from high speed flight will undoubtedly influence future research on lines similar to those indicated in the text.

In the non-metallic field, it appears that few problems will arise at speeds up to Mach 2·2. For speeds of Mach 2·7 or above, very serious difficulties are envisaged; their solution will almost certainly require the development of new materials of types different from those now used. There is a potential future for glass and ceramics.

Propulsion—The Engine Designer's View

BY

H. PEARSON, B.A., F.R.Ae.S, M.I.Mech.E.

(*Chief Engineer (Performance and Research), Rolls-Royce Ltd.*)

AN IMPORTANT POINT to realise in the question of propulsion for aircraft of Mach 2 and upwards is that efficiency of the power plant is no longer a problem. Due to the high pressure developed from the intake the overall pressure ratio of the engine is high, and the aircraft forward speed being high the Froude propulsive efficiency is also favourable. Fig. 1 shows, in fact, that at Mach 2 overall thermal efficiency as high as 35 per cent is achieved which, of course, compares favourably with the best electrical power station generation. Indeed, it is one of the main reasons why economic transport at high Mach numbers is possible. The falling aerodynamic efficiency, as defined by lift/drag ratio, is offset by this rising propulsive efficiency, so that a somewhat similar overall performance is obtained as compared with subsonic aircraft. This point has been somewhat obscured in certain quarters by considering the term LV/D in the Breguet range equation

$$R = \frac{L}{D} \cdot \frac{V}{\sigma} \log \frac{W_F}{W_E}$$

where L/D is the aircraft lift/drag ratio

V is the aircraft speed

σ engine specific fuel consumption

W_F aircraft weight at start of cruise

W_E aircraft weight at end of cruise,

as representing some form of aerodynamic efficiency of the aircraft, whereas actually it would be more instructive to consider the term σ/V as a term representing the engine specific fuel consumption on a power basis. The engine specific fuel consumption on a thrust basis, which is the one normally thought of for jet engines, does not have a very good value at Mach 2, being some 40 per cent or so higher than for subsonic aircraft, but the genuine specific fuel consumption of the engine, that based on the useful power developed, is actually much superior, as shown by Fig. 1.

Due to the high drag of power plants at high Mach number it is extremely important to get the maximum amount of thrust from a given frontal area of power plant and also, because of the high thrust necessary to propel the aircraft, a low specific engine weight is most important. For these two reasons it is desirable to operate the jet engine at as high a temperature as possible and, due to the high propulsive efficiency, there is little disadvantage in engine efficiency in doing this. Current thinking on engines for high Mach numbers designed for use in, say, eight to ten years time, generally assumes a cruising temperature in the region of 1,350°K. This is actually a figure equal to the highest temperatures being used for take-off today, so that the problem for the future in this field is the development of materials and cooling systems that will enable today's take-off temperatures to be used continuously for cruising. In this respect the main problem is the development of turbine blade alloys which will be some 100°C better than current material. The prospects of this can perhaps best be appreciated from Fig. 2, which indicates the progress that has been achieved over the years in the strength of turbine blade materials. It is considered that such temperatures, by the late 1960s, should be attainable, but it is important to realise that the onus of this development lies to a large extent with the metallurgist. The cooling of the turbine blades by compressed air from the compressor is hindered by the fact that at these high speeds this air is already rather hot, but in this respect, due to the lower engine compression ratio, the final temperature of the compressor air at cruise is rather similar to that of modern high pressure ratio engines of today at take-off, and therefore the cooling problem of the blades represents a rather similar problem to that existing on today's engines.

The intake temperature to the engine at Mach 2·2 is about 150°C, but the temperature at delivery from the compressor is still similar to that of current high pressure ratio engines. The problem of materials, therefore, for the compressor is largely that already encountered at the rear of present-day compressors. This means that more of the blades and discs have to be in steel or titanium and less in aluminium than current

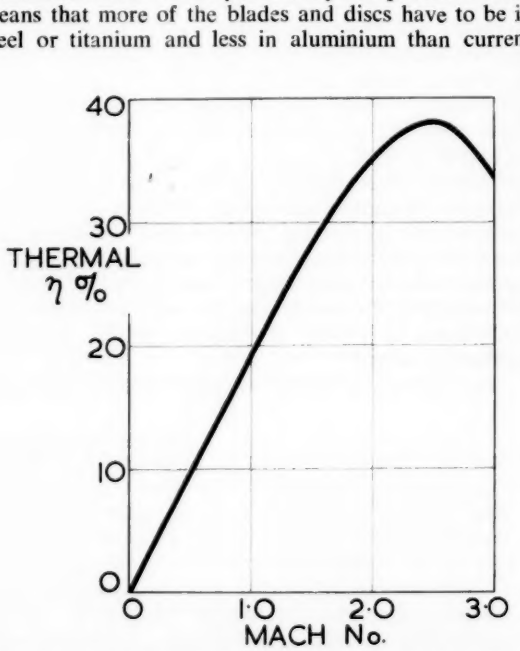

FIGURE 1. Overall thermal efficiency of non-reheat turbo-jet engine with convergent-divergent nozzle.

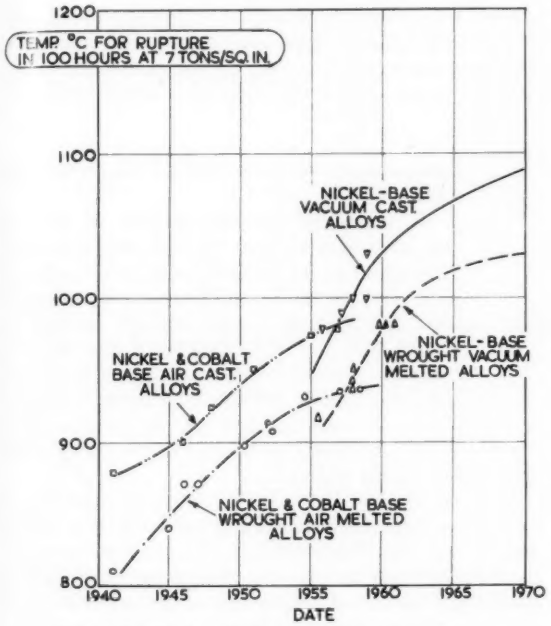

FIGURE 2. Turbine blade operating temperatures.

engines. Considerations of weight therefore emphasise the importance of using high grade material, especially for discs, to keep the weight of the engines down to the minimum.

The normal arrangements for the cooling of bearings and so on within the engine are to tap air off a relatively low pressure stage in the engine and circulate this air past bearing housings. The minimum temperature for this in the cruise condition will be about 220°C for a Mach 2·2 aeroplane. Having regard to the predicted advances in bearing materials and lubricants and to the design measures taken to ensure isolation from hot seal leakage air, it is considered that this problem is not too serious on a Mach 2·2 engine, but would undoubtedly become impossible by these means on a higher Mach number design.

The air intake is undoubtedly one of the most critical components of the engine because the pressure ratio obtained from it is rather greater than that given by the main engine compressor. The problem is largely one of correct matching between the engine demand and that provided by the intake over the range of flight conditions. Fig. 3 shows the intake critical flow compared with engine demand for a fixed intake and one in which a variable wedge is used. Using a fixed intake it will be seen that in order that the intake should pass enough flow at a low flight speed a considerable spillage of air takes place at high speed and this creates high spillage drag at the design speed. Using a variable wedge intake whose angle can be varied to give good intake pressure recovery over a wide range of speed, it will be seen that if the intake critical flow is matched to the cruise condition then there will be an excess of air at lower flight Mach numbers, which will involve spillage under these conditions. The sort of intakes that are envisaged for a transport aircraft, therefore, consist of a centrebody intake, preferably two-dimensional, whose wedge angle can be altered to suit flight conditions, and that also has spill valves fitted to duct air overboard and past the engines for low flight Mach numbers, and to ensure that the intake starts satisfactorily. The control of such a variable intake, such that it automatically matches these requirements and also that will satisfactorily re-adjust itself to prevent such things as intake buzz or instability is obviously a major development, and the control system of the power plant is a major part of the power plant design. Fig. 4 shows diagrammatically the sort of intake that is envisaged and the control system functions that have to be carried out.

The expansion ratio across the engine final nozzle is of the order of 12:1 at flight Mach number of 2 and substantial gains in performance are obtained by efficient supersonic expansion. In principle, therefore, a convergent-divergent nozzle is required for the cruise condition. However, at lower aircraft speeds and in particular for take-off, a pressure ratio of about $2\frac{1}{2}$ is all that is available, and for this pressure ratio such a convergent-divergent nozzle would give large losses and in any case would probably become unstable and exert large fluctuating side-loads on the engine. Some form of variable nozzle, therefore, is required. Various proposals have been made for such nozzles, but it has been found that they all tend to give either large losses of thrust or equivalent increases in drag in the low sonic speed range. The fundamental reason for this is that at the design condition the expanding gases occupy a

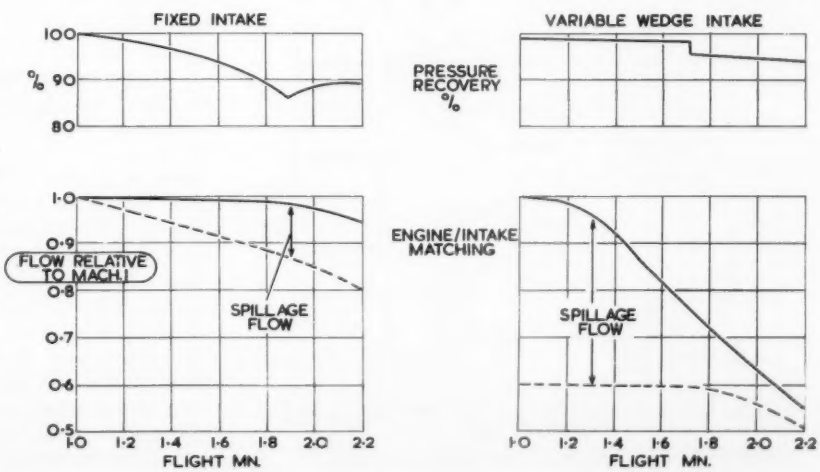

FIGURE 3. Engine/intake matching.

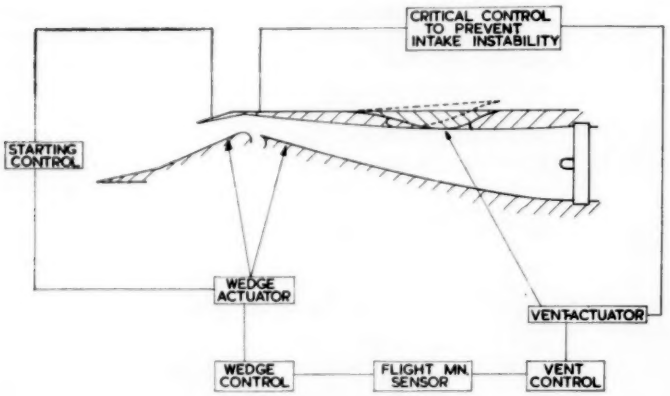

FIGURE 4. Supersonic aircraft intake control.

much larger rear end area than they do in the take-off or at low Mach number.

The rear end of the power plant, in fact, represents a critical design problem in the transonic flight phase and it cannot be claimed that a satisfactory solution has yet been reached. The claims of the ejector type nozzle need thorough exploration, particularly as in this condition, as mentioned previously, the intakes are working in the condition in which air is being spilled. It would at first sight appear attractive to duct this spillage air back past the engines to feed an ejector nozzle whose area is roughly that of the maximum of the power plant. However, ducting of this air at intake pressure past the engines is very difficult without increasing the basic diameter of the power plant itself.

It is possible that this problem of the rear end will be improved by the fact that more transonic thrust will eventually be needed in order that the transition from subsonic to supersonic speed of the aircraft shall be carried out at a sufficiently high altitude. This point is referred to later, but if the increased thrust for this condition is provided by some degree of reheat, then automatically the nozzle size of the engines will be increased and thus the base drag of the power plant will be improved.

The question of the altitude at which transition to supersonic flight takes place is one of the most critical design areas, both affecting aircraft and power plant. As is fairly well known, the altitude at which supersonic flight can take place over inhabited areas is largely governed by the intensity of the sonic boom produced at ground level. For a given aircraft design, the higher the transition the higher will be the drag of the aircraft and the lower the thrust output of the engines. This point is illustrated by Fig. 5, which shows the drag and thrust at these transition Mach numbers for different altitudes. Already current thought that this altitude will be approximately 40,000 ft. means that either the engine size is governed by this flight condition and heavy weight penalty has to be borne or some degree of reheat has to be fitted, involving its complications to engine and installation.

Lastly, of great importance for this project is the noise level at take-off and landing. For landing, as the engine is already provided with a variable intake, it may be feasible to use this to "choke" the intake and thus largely prevent compressor noise escaping. There will remain noise escaping from the jet pipe (jet noise itself is not very important at the landing condition); in this field much work is proceeding on current engines and probably by the date of this aircraft satisfactory noise levels will be obtained.

At take-off the aircraft has a large amount of power installed (some 40-50 per cent thrust/weight ratio), thus it might be thought that the aircraft would have such a rapid rate of climb that it would be sufficiently high before the airport community was reached that the noise level would be acceptable. However, climbing angle depends on maximum L/D as well as thrust/weight ratio and this is rather low for the supersonic shape chosen for the aircraft; so in this respect the aircraft is not as silent as might be thought. Moreover, such high installed thrust leads to high noise on the airport and during the early phase of the take-off. In fact, noise level becomes one of the design criteria for the engine. On present estimates it is considered that during the "fly-over" cruise, four miles from the start of aircraft roll, the noise on the ground will be approximately the

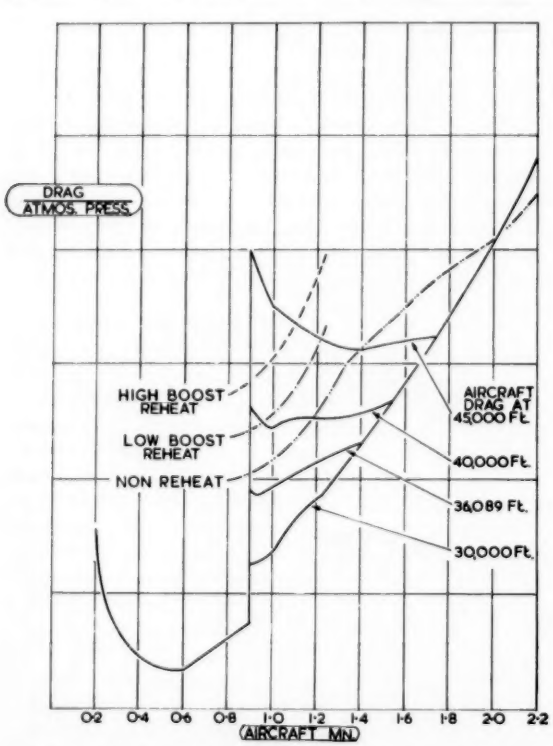

FIGURE 5. Supersonic aircraft transonic drag and thrust.

same as the current 707-420 aircraft fitted with Conways (about 102 PN dB) and the airfield noise itself will be some 3 PN dB higher. This assumes no silencers fitted to the jets as it is rather difficult to envisage such devices fitted to supersonic nozzles without prohibitive losses. But in this field it is possible that some great strides will be made before the Supersonic Air Liner takes the air.

Engines for a Mach 2·0 Transport

BY

A. V. N. REED

(Deputy Chief Project Engineer, Bristol Siddeley Engines Ltd.)

THE PROBLEM of determining the type of engine which best suits a particular application is not new to the engine designer, but the supersonic transport is perhaps unique in that a number of design criteria have been introduced which cannot easily be taken account of in the usual terms of engine weight, fuel consumption or even aircraft direct operating cost.

However, an examination of the effect of these standard criteria on different types of engines is a starting point in determining what engine will best match the supersonic transport requirement. A survey of a series of thermodynamic cycles both at Mach numbers in the region of 2·0 and at subsonic speeds typical of stand-off or diversion suggests that two types of engine merit further study. Fig. 1 shows thrust plotted against specific fuel consumption at $M=2{\cdot}0$ and $M=0{\cdot}9$ at 36,000 ft. for a turbo-jet and for a ducted fan engine of by-pass ratio of 0·7. It will be seen that for both flight conditions the ducted fan engine shows a small improvement in s.f.c. compared with the turbo-jet. This advantage will be offset by the fact that the ducted fan engine must swallow approximately 70 per cent more air flow than the turbo-jet and, hence, although the bare engine weights of the two proposals will be similar the intake ducting and final nozzle assembly weights will be considerably larger for the ducted fan engine—particularly if an integrated configuration is considered.

Figure 2 shows the weight breakdown comparing two aircraft each designed to carry the same payload the same distance, one powered by turbo-jets and one powered by ducted fan engines. The figure shows that the specific fuel consumption advantage of the ducted fan is more than offset in the real case by its increased installation weight, the ducted fan engined aircraft being some 4½ per cent heavier than the turbo-jet engined aircraft. It would be expected that the heavier aircraft would be more expensive to operate and Fig. 3 shows that in fact the ducted fan engined aircraft is some 5 per cent more expensive to operate than the turbo-jet engined aircraft. The direct operating costs have been calculated using a B.O.A.C. method and assuming that the airframe cost (less engine and power plant) was £4m.

On purely economic grounds, therefore, there is no case for the ducted fan engine, but the ducted fan

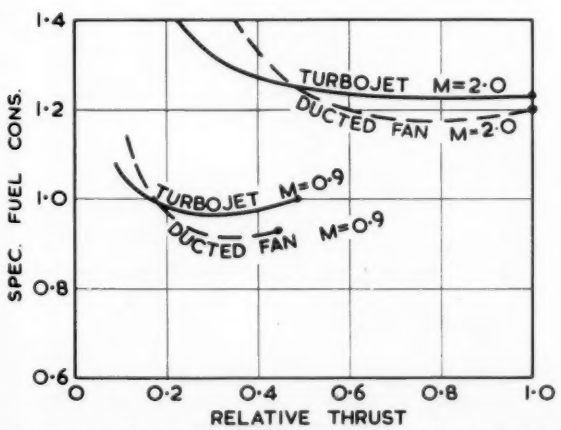

FIGURE 1. Turbo-jet and ducted fan engines: Thrust plotted against specific fuel consumption tropopause.

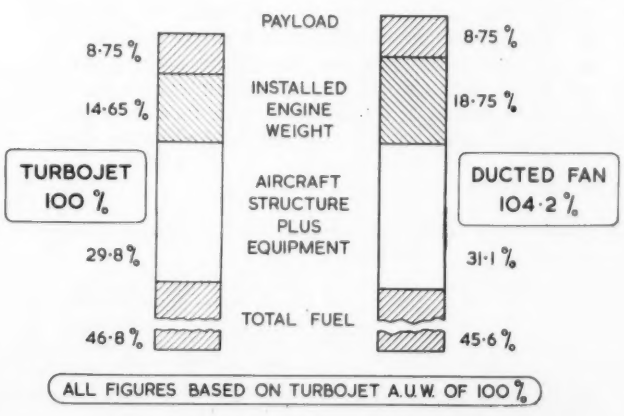

FIGURE 2. Weight breakdown for $M=2$ transports.

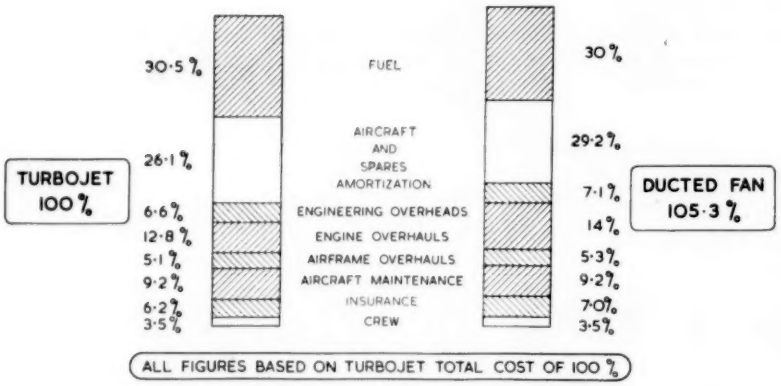

FIGURE 3. Direct operating cost comparison.

engine does offer the possibility of noise reduction. Table I shows some comparative airfield and fly-over noise estimates for the Supersonic Transport compared with a Boeing 707/320. It will be seen that the turbo-jet engined aircraft can be made some 10 PN dB quieter than the 707 at a point 4·2 miles from the start of the take-off roll. This is due to a combination of three factors:—

(*i*) the supersonic aircraft has a very much higher installed thrust than the Boeing 707, which enables the aircraft to climb more rapidly and thus achieve a greater altitude over any critical area,

(*ii*) the use of special climb techniques on the aircraft combined with reductions of thrust over critical areas,

(*iii*) the use of variable nozzle techniques on the engine to reduce thrust but to maintain mass flow.

The contribution to noise reduction of each of these factors is approximately equal.

The high thrust/weight ratio which is an advantage as far as fly-over noise is concerned is a disadvantage when airfield noise is considered—the figure shows that on the basis of the eighth power law the supersonic transport would be 6 SPL dB noisier on the airfield than the Boeing 707. This situation may be alleviated slightly as there is some evidence to suggest that the eighth power law may not apply to engines of very high jet velocity such as the supersonic transport engine. It is, however, fairly clear that the airfield noise problem will be somewhat worse than at present and that if the economic penalty is deemed acceptable then the ducted fan engine does offer a possibility of reducing noise levels on the airfield to today's values. On the other hand it may be felt that sound-proofing of airport buildings and so on is a more reasonable way of applying the penalty—particularly if they have to be sound-proofed anyway because other countries' supersonic transport designers may not give such consideration to noise reduction of the aircraft itself—and hence these aircraft may be cheaper to operate.

TABLE I

NOISE LEVELS OF $M=2$ SUPERSONIC AIRCRAFT COMPARED WITH BOEING 707 AIRCRAFT

	Peak noise at 500 ft. radius from static aircraft SPL dB	*Throttled noise level at 4·2 mile point* PN dB
Boeing 707 aircraft	Datum	Datum
Turbo-jet powered aircraft	+6	−10
Ducted fan powered aircraft	0	−19

Another noise problem which also gives cause for some concern is that associated with the sonic boom—broadly speaking, the higher the altitude at which the transonic acceleration takes place the lower the magnitude of the associated pressure rise experienced on the ground—hence the more thrust available at $M=1·0$ the higher the altitude at which the aircraft may go supersonic.

Figure 4 shows the (thrust-drag) variation for the turbo-jet and the ducted fan over a range of Mach numbers. It will be seen that the (thrust-drag) margin is least at the critical ($M=1·0$) condition, although for the case considered the turbo-jet engined aircraft can accelerate through $M=1·0$ some 8,000 ft. higher than the ducted fan engined aircraft. In both the cases considered the transonic acceleration occurs below 36,000 ft. and to achieve transonic acceleration at higher altitudes than this it will be necessary to make provision for a degree of reheat with a penalty in performance, although this penalty is much smaller than if the engine size was chosen to match the transonic acceleration requirement. From the point of view of thrust boost with reheat the ducted fan engine will show to advantage because of its high mass flow.

On the other hand, there is some evidence to suggest

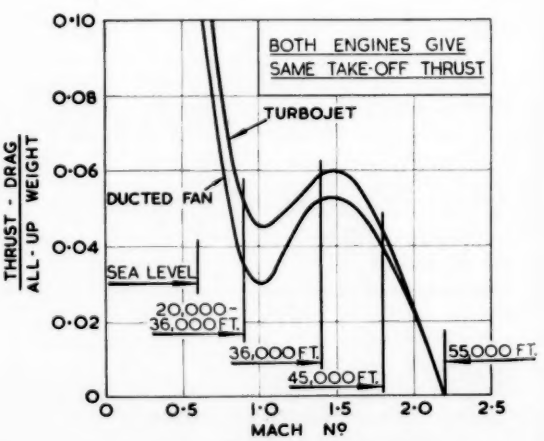

FIGURE 4. Thrust-drag margins during climb.

that in the case of a large aircraft, even if transonic acceleration is carried out at 70,000 ft., the value of the associated ground pressure rise will be approximately 2 lb/ft.^{2} While there is insufficient experience to assess how the public would react to such noises it is safe to assume that they would not like them and it may be prudent, in the first instance at least, to consider designing a supersonic transport for trans-ocean use only, in which case consideration may again have to be given to the choice of cruise Mach number.

Aerodynamics II

BY

P. L. SUTCLIFFE, B.Sc., A.F.R.Ae.S.

(Chief Technician, Advanced Projects Group, Hawker Siddeley Aviation Ltd.)

Choice of Intake-Engine-Exit System

The flight plan of the supersonic transport demands efficient aerodynamics and propulsion over a wide range of speed and altitude. As an example, the typical trans-Atlantic flight shown in Fig. 1 demonstrates that apart from cruising efficiently at, say, $M=2{\cdot}2$ for 2,500 n.m. at altitudes from 50,000 ft. to 65,000 ft., the aircraft must also be efficient at subsonic speeds. This is particularly true on the portion of the flight after the normal stage length, *i.e.* the diversion and hold, since the fuel required here must be carried the whole way and the "growth factor" on aircraft size is large.

It should be noted that about 20 per cent of the

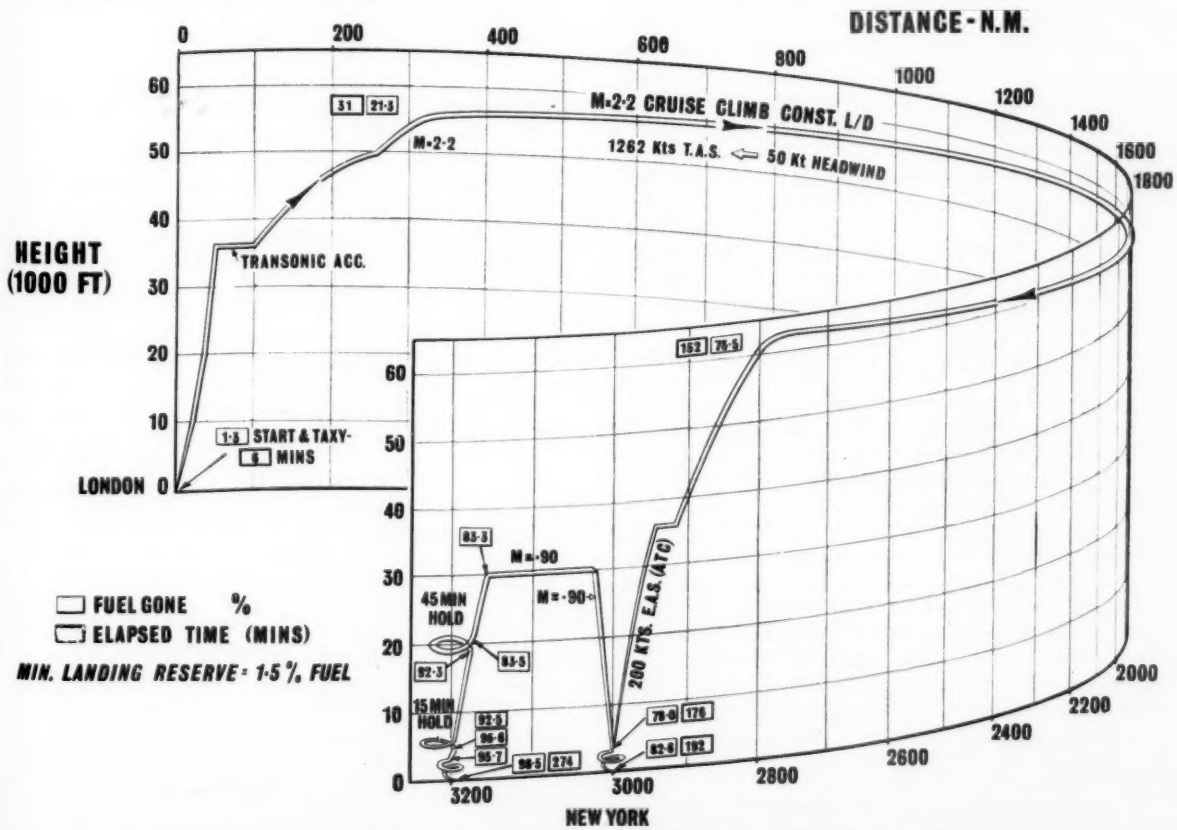

FIGURE 1. Flight plan for $M=2{\cdot}2$ supersonic transport. Non-stop trans-Atlantic.

total fuel carried is used on the subsonic climb, transonic acceleration and supersonic climb to start of cruise height, and 55 per cent on the supersonic cruise itself. The remaining 25 per cent of the fuel carried is used in the final let-down and for reserves.

The aircraft and power plant must therefore have as high an efficiency as possible at both subsonic, medium altitude and supersonic, high altitude conditions and this inevitably leads to compromise. Some of the problems of obtaining an efficient airframe under these conditions have been covered in an earlier paper, but a few notes here on the engine-intake-exit system may be of interest.

Firstly, at the supersonic cruise conditions the air intake is responsible for between half and three-quarters of the overall cycle compression and consequently the attainment of high intake pressure recoveries is just as important as high compressor efficiencies. It is hoped that pressure recoveries as high as 92 per cent at $M = 2{\cdot}2$ may be attained within the next decade with throat bleed penalties below 5 per cent of entry mass flow. Such an intake will have variable geometry, boundary layer control, auxiliary entry and/or bleed doors.

Secondly, since the expansion ratio across the propelling nozzle is from four to five times that across the turbine, and the return in net thrust for unit increase in nozzle propulsive efficiency is about 3 : 1, it is essential to have as efficient a final nozzle as possible. This leads to the use of variable supersonic nozzles of one form or another. Unfortunately the necessity for high nozzle efficiency makes the use of permanently attached silencers or thrust reversers most undesirable unless they can be provided for negligible penalty on cruise thrust coefficient.

The remaining item in the propulsion system is the engine itself and although it forms only part of the overall system at the cruise condition it is much more dominant on the subsonic portions of the flight. Two engine types have been studied in some detail, straight turbo-jets and by-pass or ducted fan engines. The straight turbo-jet appears to offer the least installed weight and drag of any type, providing it has acceptable noise at take-off and can provide sufficient thrust margin for the transonic acceleration.

The by-pass engine shows a better subsonic efficiency but is more bulky and thus may bring drag penalties in cruising flight. However, its noise level is less for a given installed thrust and if reheat is introduced into the by-pass air then the transonic thrust can be considerably increased for a relatively small fuel penalty. This form of mild reheat can also be used on the cruise to increase the thrust and so reduce the required engine size; if the by-pass reheat temperature is kept to the same order as that of the normal turbine outlet temperature then there need be no deterioration in engine life. Finally, the availability of reheat in the by-pass when required offers additional thrust in the engine-cut take-off case and in adverse temperature and altitude conditions.

It can be seen from the above that the ultimate choice of engine type for the supersonic transport depends on many inter-related features, not the least of which is noise at take-off, which, unfortunately, is a very ill-defined criterion to be met, particularly eight to ten years hence. However, since it appears obvious that public reaction to noise is increasing, and allowable limits may decrease rather than increase, it may be sensible to choose the engine with a noise level well below the present limitation, this would allow for more stringent regulations in the future and also for the adverse effects of weight growth and so on, in the development life of the aircraft.

Installation of Propulsive System

The installation of the intake-engine-exit system in the airframe is the problem not only of the propulsion expert, but the solution must be acceptable to the structural expert, the weights controller, the servo-mechanisms expert, the servicing engineer, etc. The original ideas on engine installation were largely based on the "buried" or "semi-buried" arrangements where the excess area of the engine plus surrounding structure over the intake area is buried within the aircraft to minimise the external wave drag. As four or six engines will be required for the supersonic transport, blocks of two or three engines lying side-by-side behind "two-dimensional" intakes were proposed (Fig. 2).

The alternative approach, so much used by the Americans, of separate pods for each engine, was not favoured initially due to the large ratio of engine plus structure area to intake capture area, and hence high wave drags. Recent work, however, has shown that the increased capture area due to higher intake efficiencies now thought possible, plus refined cowling techniques which reduce the maximum carcase area, makes the ratio of these two areas very nearly unity. Thus it now seems possible to have a podded installation without large drag penalties, in fact, our estimates for the total drag of the podded installation are almost identical to those for the semi-buried installation. The weight of the two systems is also very similar, the increase in

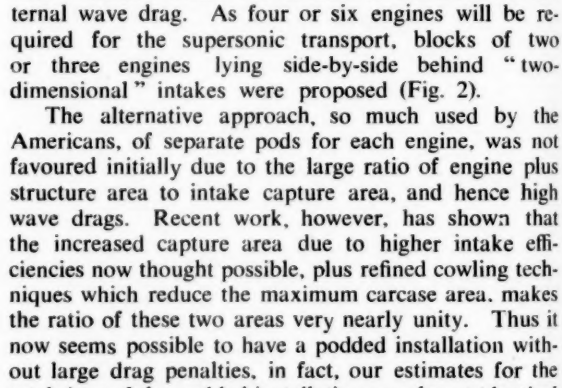

FIGURE 2.

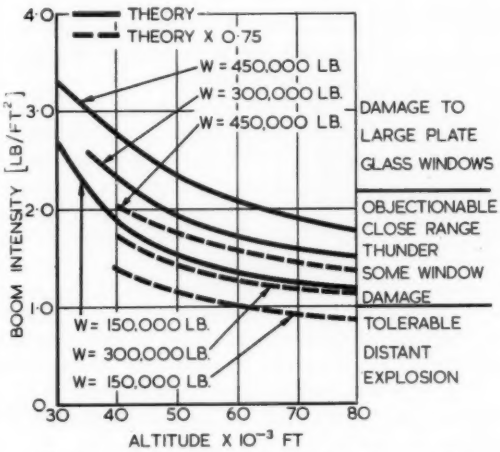

FIGURE 3. Boom due to typical $M=2$ transports.

weight due to the pylon mounting the pod to the wing being offset by the much reduced weight of the circular intake and exit as compared to the large flat moving surfaces in the semi-buried arrangement.

Two final considerations, however, appear to weigh very heavily in favour of the podded arrangement. Firstly, the biggest single problem of the semi-buried arrangement is the treatment of the aircraft boundary layer, which can be as deep as 8-10 inches at the intake ramp. Some means of removing or deflecting this boundary layer must be found and this can cause considerable penalties in weight and drag. On the podded installation, however, the only boundary layer is that on the centrebody itself and relatively simple and well-tried suction methods can be used for its removal. Secondly, there is little doubt that the accessibility of the podded installation is much superior to that of the buried arrangement and, furthermore, a change of engine type or size can be accommodated much more easily.

It appears, therefore, from the above that the podded arrangement offers a less complicated and more flexible engine installation than the semi-buried one for little or no penalties in drag and weight.

Problem of Sonic Boom

Engine noise at take-off and landing has been discussed in detail in previous papers, but another possible noise problem may be introduced with the supersonic transport. This is the sonic boom produced by any aircraft flying at supersonic speeds and, although it is not new, very few of the population of the world have been subjected to it with the regularity that will come with scheduled airline service. The boom is caused by the shock waves from the aircraft reaching the ground and above a Mach number of about 1·6 is sensitive only to two major parameters, the weight of the aircraft and the height, assuming level flight.

The full lines of Fig. 3 show the boom level in lb./ft.² against altitude for aircraft weights of 150,000 lb., 300,000 lb. and 450,000 lb., as calculated by Morris' formulae of Ref. 1, *i.e.* including lifting effects. The dashed lines show what may be the actual boom pressures produced on the ground, the 75 per cent factor being suggested by some recent unpublished American flight test and wind tunnel data. It should be noted that the pressures shown in Fig. 3 contain a reflectivity factor of 2, *i.e.* are twice the free air values, and are equivalent to what the human being would experience when standing on level ground.

On the right hand side of Fig. 3 are shown possible gradations of boom levels in terms of tolerance, annoyance, damage and so on. It must be emphasised that these levels are as yet only tentative suggestions, but authorities on both sides of the Atlantic agree that a 1 lb./ft.² boom (including a reflectivity factor of 2) may be about the maximum that the population would put up with without serious objection. This level must be treated as provisional at this early stage, but if it is justified as more experience is gained, then it could have serious repercussions on the routes that could be economically flown by supersonic transports and hence on their utilisation and their total market potential.

It appears that any supersonic transport weighing more than about 200,000 lb. will produce a boom greater than 1 lb./ft.² whatever its cruise altitude, at least up to 100,000 ft., which is considerably higher than anyone is contemplating today. Also the boom produced during transonic acceleration, where the aircraft is heaviest and the altitude fairly low, will be of even greater intensity and could cause damage.

The areas affected by the boom from a typical supersonic transport are shown in Fig. 4 (assuming the full

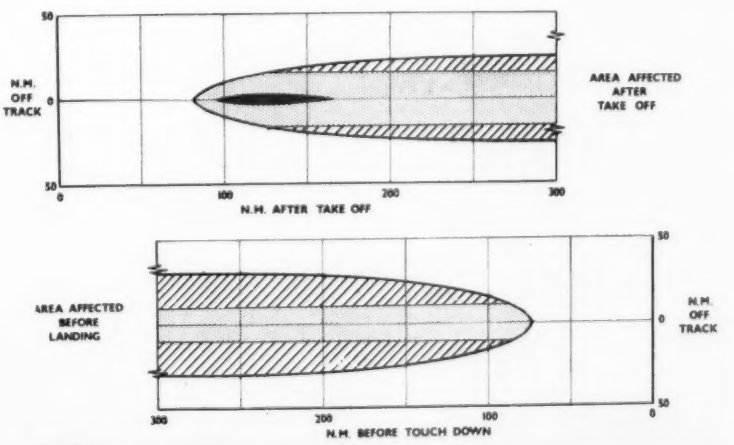

FIGURE 4. Area affected by sonic booms $M=2{\cdot}2$.

theoretical boom pressures) and it will be seen that the transonic acceleration causes a narrow band of high intensity boom about 100-150 n.m. after take-off, but the boom due to cruising flight affects an area about 30 n.m. wide under the whole of the flight path until the aircraft decelerates to subsonic speeds about 80 n.m. from destination. This latter area will be subjected to a boom of greater than 1 lb./ft.^{2}, in fact, with aircraft sizes at present being considered a typical level is about 2·0 lb./ft.^{2} (or 1·5 lb./ft.^{2} with the 0·75 factor).

Now, considering typical routes that may be flown by supersonic transports, while there is probably little to worry about on the London-New York run, there may be great problems on the South African, Far Eastern and Australian routes. Furthermore, there must be very great doubt whether a supersonic transport could be used on the American coast-to-coast or domestic network at all.

If the supersonic transport is not allowed to fly at supersonic speeds over populated territory, then its block time will increase and this, plus the increased fuel required, means a considerable increase in direct operating cost. If the block time on mixed over-land and over-ocean routes approaches that of conventional subsonic jets then the justification for the supersonic transport on these particular routes is much reduced and so the possible market potential falls also.

It is suggested that this may be one of the most important aspects of the operation of the supersonic transport and may be a deciding factor in its eventual commercial acceptance by the airlines. It is essential, therefore, that much more supersonic aircraft flying experience must be obtained in the very near future, using controlled experiments from which it should be possible to determine the level of sonic boom that will be acceptable to the general public. Only then will it be possible to assess the overall sales potential of the supersonic transport, but there is no doubt that in the meantime we must continue our studies into the problems of designing, developing and finally manufacturing these aircraft.

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1. MORRIS, J. (1960). An Investigation of Lifting Effects on the Intensity of Sonic Booms. *Journal of the Royal Aeronautical Society*, October 1960.

Airworthiness

BY

D. M. JAMESON, M.B.E., A.F.R.Ae.S.
(Advanced Project Group, Air Registration Board)

1. Introduction

I would like to preface my remarks by gratefully acknowledging the inspiration I have received from my colleagues and others with whom the problems I shall talk about have been discussed, and by making it clear that, although my views are naturally coloured by the work on which I am engaged, they must not necessarily be taken as the official views of the Air Registration Board.

The supersonic aeroplane, due to its novelty, presents the three main types of airworthiness problem in a more pressing form. These are: knowledge of the environment; knowledge of what the aeroplane must be able to do in that environment (which includes a clear idea of acceptable risk); and obtaining, before passenger service, the necessary confidence that the intended safety level will be achieved. I can only touch on a few of the individual problems.

2. Environment

2.1. GENERAL

Starting with environment, we need data on the atmosphere, not only for structural, but also for handling and power plant requirements. It must cover an extended altitude band, be in a pure form, *i.e.* not modified or selected by the response characteristics of the measuring vehicle, and be comprehensive. For example, it must contain data on wind shear and temperature gradients (both important for speed margins), and on hail.

2.2. HAIL

In Fig. 1 I have reproduced, by courtesy of the U.S. Military Air Transport Service, a diagram of USAF in-flight hail encounters. This diagram does *not* show the relative probability per flying hour of meeting hail at different altitudes, but it does indicate the need to agree a hail spectrum for design purposes. I have inset a photograph illustrating the damage a 3-inch hailstone can do to a *present-day* aeroplane. The supersonic aeroplane is less capable of slowing down or taking evasive action.

3. Ability

3.1. AIRWORTHINESS ENVELOPES

In its environment, what must the aeroplane be able to do? Analysis of the range of intended flight plans—one of which is illustrated in Fig. 2—will give for each

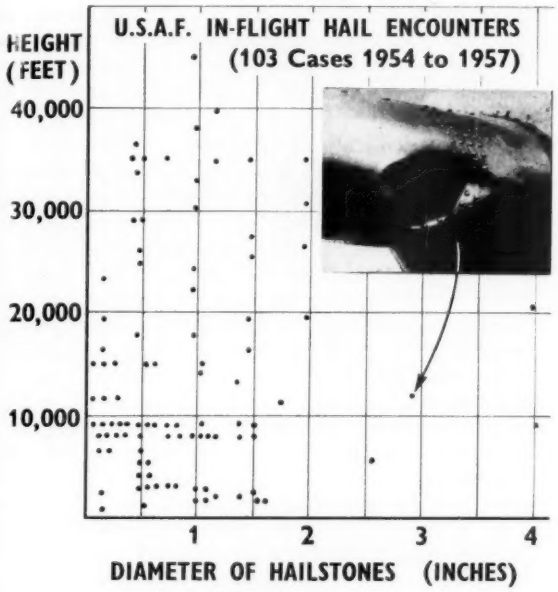

FIGURE 1.

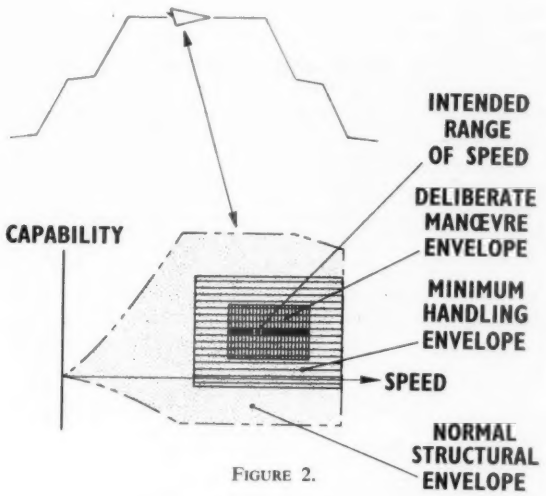

FIGURE 2.

altitude a range of intended operating speeds (shown as a heavy line). Over this range of speeds, chosen by the designer, the pilot will want a certain capability for deliberate manœuvre. Outside this again is the minimum handling envelope, which provides for inadvertent excursions. Handling qualities for this " excursion " envelope should be specified in the requirements and demonstrated in flight. The minimum envelope for structure, and for functioning of non-interruptable services, must be outside this by an amount which could be affected by any feature which can act as a reliable manœuvre-limiting device. Between the handling envelope and the minimum structural and functioning envelope there shou!d be no reason to suspect a catastrophic deterioration of handling qualities. These airworthiness envelopes have to be determined, but at least the problem of defining and using such terms as " stalling speed " should be largely avoided.

3.2. COMPLEX LOADING CASES

Returning specifically to structures for a moment, there is the need for examining additional complex loading cases, for example, the "quench" case, where the aeroplane enters turbulent cloud while still hot, thus encountering both gust and thermal shock loads while the material properties are still degraded by heat.

3.3. THE "HIGH PROOF CONDITION"

In evolving appropriate structural cases we shall, most probably, have to consider the "high proof condition," *i.e.* that condition between the present proof and ultimate, which the aeroplane can withstand just sufficiently well to be able to complete its flight given no further major emergencies.

3.4. GROUND CLEARANCE

Figure 3 (which does not portray an "optimised"

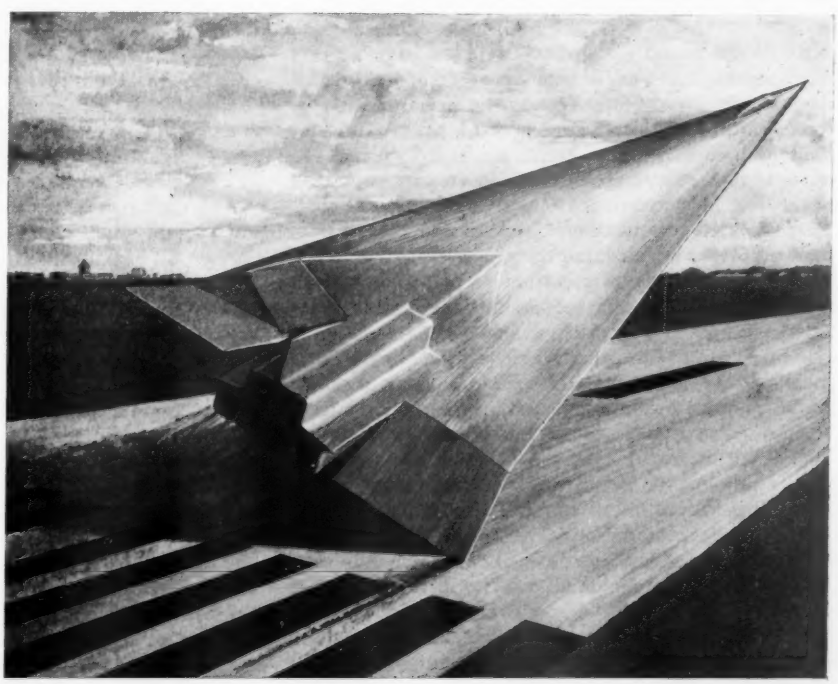

FIGURE 3. Ground clearance.

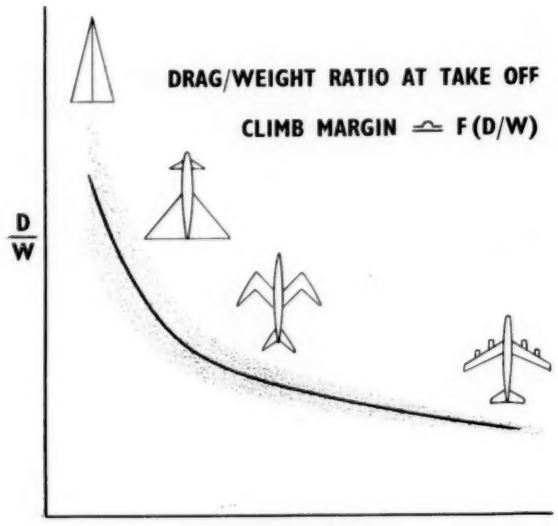

FIGURE 4.

design) shows that new emphasis is given to the ground clearance problem. Scheduled field lengths are sensitive to unstick and touch-down speeds, which must in turn be suitably separated from the *minimum possible* unstick and touch-down speeds in maximum cross-wind conditions. These minimum speeds may well be limited by ground clearance, which must allow for all reasonably possible combinations of bank, pitch, undercarriage deflection and control deflection. If the aeroplane is likely to overlap the edge of the runway, snowbanks and runway edge lights on their frangible stalks should be considered also.

3.5. AIRFIELD AND CLIMB PERFORMANCE

Figure 4 illustrates for low speed flight the trend of drag/weight ratio versus aspect ratio. Although drag/weight ratio is by no means the only parameter determining climb margins and manœuvre performance, for conventional aircraft these have been expressed with reasonable accuracy as straight multiples of D/W. From a brief examination it appears that, for a given number of power units, accuracy of control and so on, the required manœuvre performance plus climb margin for supersonic aircraft increases *more* rapidly than D/W. Thus, to a take-off second segment datum of 2 per cent (and to the effective obstacle clearance slope) must be added roughly a 2 per cent gradient for an M-wing aircraft or a 4 per cent gradient for a slender delta, in place of the one per cent gradient applicable to present 4-engined aircraft. This example is chosen to show the order of problem before us. However, all field length and climb cases need examination, including those at present covered indirectly like the two-out approach and one-out balked landing. It seems likely that the margins finally used will have to be derived for each particular design. Initially, fair estimates will be necessary, but route proving must provide data enough for at least conservative margins to be usable with confidence when fare-paying passengers are first carried. Thorough recording and analysis of flight experience should then permit margins to be safely reduced to the design values, when these are substantiated.

4. Establishing Confidence

For a novel aircraft intended to cruise in a relatively unexplored environment, and more dependent than current aircraft on an array of sophisticated systems, it is hoped that full use would be made of multi-purpose ground testing rigs and realistic simulators to reveal many of the otherwise unsuspected problems and thus enable the necessary reliability and consistency of systems, piloting techniques and so on, to be developed before the aeroplane ever flies. However, practical flight testing and operational experience (both flying and maintenance) will be needed, and will need to be unusually comprehensive and well instrumented. It is impossible, of course, to guarantee that any particular number of hours or flights will be sufficient but, as regards route proving, it would be well to allow for, say, at least one calendar year of operation, embracing:—

(*i*) exposure to a fair sample of operators' personnel, facilities and procedures,

(*ii*) experience of a wide range of climatic conditions during both flying and maintenance,

(*iii*) maintenance of several individual aeroplanes over one full maintenance cycle, and of one or two aeroplanes over two or more full maintenance cycles,

(*iv*) collection of valid statistical records sufficient to support initial introduction into service.

Item (*iv*) should confirm estimates of, for example, load and temperature spectra, handling and performance variations during take-off and landing, and the behaviour of power units and of auxiliary systems. As regards Items (*iii*) and (*iv*) there is some scope for shortening the proving time if initially conservative margins for

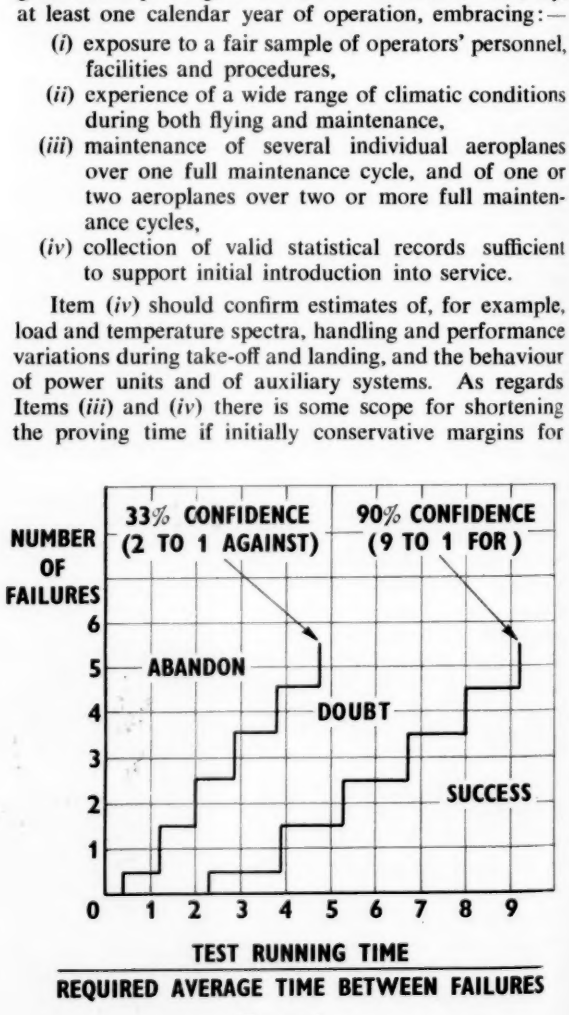

FIGURE 5.

passenger service are accepted. I am thinking here of items such as:

climb and field length performance
cross-wind capability
cruising speed
fuel reserves
weather minima
intervals between inspections or replacements.

Figure 5 relates to any reliability measurements on ground rigs or in proving flying where the failure pattern is substantially random (as is often the case). It shows how the observed failure rate affects the experience required to establish, at a conventional confidence level, that a given target reliability has been met. It will thus be seen that proving flight time, and some ground rig testing time, will be affected by how well the aeroplane behaves relative to its design targets, which themselves may be modest or ambitious.

I would emphasise that it is not my aim to argue numbers here (though I hope they are of the right order) but to illustrate some of the considerations underlying the choice of proving programme. This is a matter of some importance, since unless full advantage is taken of all possible methods of alleviation, it may well be found that something in the order of 15,000 hours flying between first flight and passenger service is needed to establish the customary confidence that the normal safety level will be achieved.

Cockpit Design and Management

BY

CAPTAIN F. ORMONROYD, D.F.M., M.A.(Oxon.), F.I.N.

(Flight Technical Superintendent, Flight Operations Requirements Branch, B.E.A.)

THE RATE OF error in the performance of any task increases rapidly after a certain degree of complexity or rate of operation is exceeded. As aircraft speeds increase, therefore, the safety and efficiency of the operation will depend largely on the extent to which the job of the crew can be simplified and on the work-load being kept within tolerable limits. In recent years the problem has been tackled by the addition of specialist crew members and by the use of automatic and semi-automatic equipment. Distribution of the total load over an increased number of people is no longer a satisfactory solution, however, since it imposes an additional burden on the Captain in his capacity as overall monitor and the time available for this vital task is all too little. While the pilot by tradition tends to reserve judgment on automatic devices generally, where a high standard of reliability is proved the system will receive acceptance in due course and there is little doubt that it is in this field where the greatest contribution to operating efficiency will be found.

What, then, will we expect to find in supersonic transport aircraft? An auto-pilot with an exceptionally high standard of reliability and possessing much greater authority than hitherto will undoubtedly be required and this will mean a multiplex or equivalent system to provide the necessary safeguards. In addition to the usual manometric locks (I.A.S., Mach and height) a facility for pre-selecting height would be of immense help to the pilot, not only as an " aide memoir " but because of the very real danger with high rates of descent of passing through the altitude at which he is required to fly level. The approach characteristics of the aircraft will almost certainly demand automatic speed control (with or without the auto-pilot in use) and the final landing must be capable of being carried out in the worst weather conditions, in which case automatic control, including flare-out and drift kick-off just before touch-down, is bound to be specified.

In the semi-automatic field the pilot is fairly certain to require director guidance during the take-off run in poor visibility conditions and during the rotation and initial climb, and one of the new " head-up " displays will probably be used for this purpose. In the realm of navigation aids experience has shown that a map-type of display is not only useful as a " situation " monitor, but also enables the pilot to maintain a predetermined track or holding pattern to a very high order of accuracy and the use of this type of aid is expected to increase. The development of an automatic signalling system between the ground and the aircraft is being progressed and it is to be hoped that the system will be available in time since communication by voice is barely satisfactory even today.

The increasing use of automatic systems invariably results in yet more warning lights to add to the already considerable number, and a centralised warning system employing flashing master warning lights to be cancelled by the pilot and a display panel on which individual system windows (*e.g.* ENG 1, HYDRAULICS etc.) remain illuminated until reset, is a long-awaited development.

We must not omit other less obvious but equally important ways of helping the pilot to do a good job. Window frames symmetrically disposed about the forward line of vision and a horizontal coaming cut-off line, for instance, can be invaluable when assessing the

attitude and flight path against a background of approach lights in poor visibility. The downward view on the approach must be at least 10° below the horizon to obtain satisfactory visual guidance under such conditions and this means 20° to 25° relative to the fore and aft axis. Although this figure is half as much again as that currently available, the continuing trend towards integration of the flight instruments should result in a reduced demand for panel space. The screening of instruments by the control wheel, temporarily or otherwise, cannot be tolerated, however, and a fresh approach to this problem may lie in substituting a small push-pull tiller for the conventional " stick."

But most important of all is attention to design, layout and method of operation of all systems, controls and displays. Time does not permit a discussion of details, but it must be mentioned that much published work exists on this subject, work which, unfortunately, has been ignored in the past but the results of which are well worth applying.

What crew complement will be carried and what sort of operating technique adopted? Because of the reduced block times the operation could well be similar to that of a short-haul subsonic jet aircraft flown by, say, a crew of three pilots. The aircraft could be so designed that the actions required could all be carried out by any two of the three pilots in any seat position under normal operating conditions, leaving the third pilot available to monitor their actions and give assistance in emergency if required.

A monitored approach procedure whereby the second pilot flies the aircraft and the first pilot monitors the situation, has been a feature of B.E.A.'s Viscount operations for some time now with very satisfactory results. With a two-pilot operation, however, it is inevitable that occasions arise when the first pilot carries out drills which cannot be cross-checked by the second pilot, who may be fully occupied flying the aircraft. It is in situations such as these where the " onlooker sees most of the game " and where a third member with little or no physical duties to perform is more likely to detect an error and stave off a serious accident. Because of " biased probability," however, we cannot expect the procedure to be completely successful. The ability of a human being to detect error or an isolated abnormal case on, say, one per cent of occasions would appear to be seriously reduced by " mental inertia " resulting from the knowledge of normal operation on the other 99 per cent.

As automatic monitoring systems are not affected by " biased probability " and as the repeated performance of sets of actions (whether in sequence or not) are more reliably done by automatic means the ideal arrangement would appear to be one automatic system checking the actions of another. The triplex auto-landing system is, perhaps, a good example. The next best approach is to let the automatic system monitor the man. We are all familiar with the horn which blows when the throttles are closed and the undercarriage has not been lowered. Unfortunately the warning usually occurs at much too late a stage, but this does not detract from the soundness of the principle!

Engineering Systems on Supersonic Aircraft

BY

R. C. MORGAN, O.B.E., F.R.Ae.S.

(*Chief Project and Development Engineer, B.E.A.*)

A GREAT DEAL has been said and written about the necessity to launch into a completely new standard plateau of systems engineering for supersonic, as distinct from subsonic, aircraft. So far as I have been able to gather, one reason for all this could be the argument that, as the supersonic aircraft will be so costly to operate, *i.e.* so uneconomical, it must be operated at a very high utilisation with very short turn-rounds and that therefore all the systems must be very much more durable than has previously been the case. Another reason could well be that the flight régimes are such that there will have to be much more dependence on certain systems, *e.g.* pressurisation at 70,000 ft., cabin cooling at very high speed and so on.

I wonder whether all this is logical. In the first place, what is the difference between a supersonic aircraft and a subsonic aircraft when it comes to the question of operating at a very high utilisation to get the costs down? If the supersonic aircraft is very expensive to operate, then it should not be bought at all and, on the other hand, if it is possible to operate at very great utilisation with short turn-rounds, then this should be done with the subsonic aircraft, and I am forced to ask the question, " Why is it not done?" Surely the answer cannot be simply, " Just because of the standard of systems engineering," so what are we talking about?

To try to get down to it, and taking the second point first, it is obvious that, for the more difficult flight régimes of the supersonic aircraft, there will be a higher degree of dependence on certain systems than is the

case at present, so the first thing to do is to find out which and under what circumstances and then to decide how this dependence is to be satisfied. I cannot help thinking that, while every effort has got to be made towards greater reliability, and there is a tremendous amount to do here on our current subsonic aircraft, we shall always finish up with the situation that, where dependence (safety) is concerned, system multiplication of one kind or another will be essential.

With regard to the point about high utilisation, perhaps in the discussion someone can explain why it is apparently considered that, from a reliability point of view, systems engineering (and I will put the question round the other way) can be allowed to be worse on subsonic aircraft than on supersonic aircraft? The short-haul operator already has the problems of maintaining high utilisation when there are difficulties of integration of services and of attracting passengers to fly when he wants them to fly. So the pressures of providing enough reliability to ensure that as much as possible of the maintenance is planned maintenance is already with him. On the other hand, the long-haul operator of subsonic aircraft, by virtue of the wide geographic scattering of his fleet, experiences the same pressures because he cannot afford to scatter his maintenance and spares facilities all over the world.

Subsonic Aircraft Systems

With these factors in mind, it is worth taking a look at where we stand in respect of systems engineering for the current and for the next generation of subsonic aircraft. The next generation is a quite sophisticated one. Take the D.H. Trident for example.

FLIGHT INSTRUMENTATION AND AUTO-PILOT

We have a fully integrated instrumentation and automatic control system. The flight instruments, flight director and auto-pilot take electrical signals from shared central data sources. These sources, which are duplicated or triplicated for safety reasons, consist of high accuracy gyros and air data computers. The flight director and auto-pilot provide compatible flight control facilities, including automatic flare-out for landing. The auto-pilot is initially duplicated, but can be triplicated for automatic landing.

CONTROLS

Apart from the conventional controls, we have droop leading edges, lift dumpers, air brakes and divided ailerons. All are hydraulically operated by a triplicated system.

ELECTRICAL SYSTEMS

The 70 KVA 400 cycle generating system is orthodox. An interesting feature is the central warning system. Warning indications at the systems panel (adjacent to operating switches) are brought to the attention of the pilot by a flashing light in front of him and by illumination of a window which indicates which system is faulty. The flashing light can be cancelled and is then available for any other fault that may develop in the aircraft systems. In the case of a " shut-down " warning, a lamp in the throttle lever of the engine concerned also lights up.

AIR CONDITIONING

The cabin is pressurised to $8\frac{1}{4}$ lb./in.^{2} by duplicated air bleeds from the engines. Temperature control is achieved by compressor heating and air cycle cooling.

OTHER AIR BLEEDS

Engine air bleeds are used for airframe and engine anti-icing.

RADIO AND RADAR

Functionally, the airborne equipment enables air-to-ground communication by conventional means, with special provisions for future implementation of automatic communications systems and for receiving teletyped messages as, for example, Met. reports. There are receivers for ground-based position-fixing information, including VOR/DME, Decca and ILS. The Doppler radar is independent of ground stations and provides ground speed and drift to a navigation computer, which processes this information to display present position continuously in a map display. Weather radar is also carried. The radio and radar equipment is extensively transistorised.

Supersonic Aircraft Systems

For all the above equipment, on a new subsonic aircraft, the operational requirements are quite clearly specified, the components are made to known techniques and we are entitled to assume that the amount of " debugging " will be no more than for a comparable quantity of new equipment in the past. With this in mind, let us look at the sources of unserviceability that can lead to engineering delays on a current jet aircraft, reasonably developed, and operating to a utilisation and turn-round schedule that may well be similar to what the long-haulers want for their supersonic aircraft.

The rate of unscheduled unserviceability is, we must admit, pretty high at about 50 hours per 100 flying hours. Of this, 11 hours is due to the items that we might anticipate being doubled up in complexity and/or quantity for a supersonic aircraft, namely, air conditioning, radio, flight systems and auto-pilot. Allowing only for the normal development of maintenance techniques, with increases in quantity of these named systems and of electronics in engine and fuel system controls, it does not look as if the unserviceability for a supersonic aircraft will be more than, say, 30 per cent up on existing rates. This, although significant, does not in itself call for a new order of systems engineering. To my mind, it confirms that the real new problems we have to face are not primarily those of high utilisation, but those of ensuring the safety of the aircraft because of the much greater dependence of this on certain particular systems—air conditioning and variable geometry both for airframe and engine intakes. I do not believe that the flight systems will be a very great jump from the system we are getting for the Trident, but, whereas the full auto-landing provision

on the latter will be only rarely required in earnest, simply due to poor weather, it may be essential on the supersonic aircraft in good conditions, if a pilot-judged landing is excessively difficult (calling for a lot of practice) or impossible. It is very likely that use will be made of automatic programming that will take over much of the crew's duties in respect of setting up the radio facilities or auto-pilot and for fuel control.

Air conditioning and internal temperature control appear to dominate all other systems in the supersonic aircraft. In the high temperature environment of the supersonic régime—with skin temperatures of around 100°C at $M=2$ and 250°C at $M=3$—not only the passengers in the cabin, but also many system components, will have to be protected from this heat. Examples are: electronic components, particularly where transistorised, hydraulic components (unless there are radical developments in fluids), landing gear and fuel system. Depending on the duration and speed in the supersonic régime (which may be quite a small proportion of the flight time), some of this protection can be achieved by insulation, together with some reliance on the heat sink characteristics. But it is obvious that the temperature control to be exercised by the air conditioning system is as important as the cabin pressure control on today's 40,000 ft. jets. And, because much of the flight is in a rapidly changing temperature environment (say from 15°C at Sea Level to –25°C (indicated) during climb and then up to 100°C or even 250°C a few minutes later during cruise, and with similar changes during descent), the requirements for sensing, signalling, computing and control equipment will be extremely rigorous and will lead to a great increase in electronics and circuitry. I will not attempt to pronounce on the best type of air-conditioning system, the source of air, the cooling cycle, the proportion of re-circulated air, and so on. Absolute integrity of the pressurisation system is essential. This means not only must structural strength be assured, but also the control system must be made proof against failure and mishandling. Oxygen is of no value if the cabin pressure fails at 70,000 ft.!

Many of the problems associated with variable geometry are simply a matter of degree when compared with what is current in the way of aerodynamic devices, already triplicated on the Trident. Discussion of major developments like variable sweep-back and boundary layer control are not very valuable without a specific aircraft in mind. I will only say that, once these have been shown to be mechanically and operationally feasible, the control requirements should not be too difficult. Variable geometry for engine intakes and exhausts is another matter. Here there will be an increase in control complexity due to the inter-acting requirements of engine, inlet, fuel system, after-burner and nozzle configuration.

These then are some of the new and trying features of aircraft systems on supersonic aircraft. To ensure safety, despite all our attentions to reliability, I have no doubt that system multiplication will be a frequently-used tool. The problem is, how do we tackle the important, but not-to-be-exaggerated, potential increase in the unserviceability that can cause engineering delays?

I suggest that the line of development should be a well-balanced approach to the problem of fault diagnosis. This involves:—

(*i*) Deciding what can be expected in the way of time and technical ability of maintenance staff at and away from Base. This will determine the level of breakdown for component changes on system defects and the type of test equipment. In general, we will find that testing away from Base should be avoided and that systems should be designed to indicate clearly the component that has failed. This is one possible field for flight-recording systems.

(*ii*) Ensuring that the necessary test points and equipment are provided. An essential requirement is that, when these facilities are superimposed on a multiplex system, they do not interfere with the functional independence and integrity of the parts of a system.

It appears that most of the increasing complexity of the supersonic aircraft will be in systems whose signals or outputs are electrical functions. This is the field where fault diagnosis can be most easily developed.

On civil aircraft I do not believe there is anything to be gained by elaborate automatic check-out equipment. For safety we ensure that any system with a finite risk of failure is redundant, either because we can, if pressed, get along without it, or because it is duplicated. On an aircraft that spends much of its day in the air, the mere performance of its task is sufficient to indicate serviceability. To impose more frequent checks of most of its equipment is not necessary. In the particular case of electronic equipment we can say that the act of switching is the most likely cause of failure, whether for operation or just for test; and we should not exaggerate the need for in-flight monitoring or warning equipment in cases where we are basically relying on redundancy. Not only is the monitoring system likely to be no more reliable than the main system, but there are considerable problems attached to setting tolerances for the allowable drift before a system is quite unusable.

I have deliberately not led up to any specific conclusion because I believe the purpose of all papers at the Symposium is to stimulate thought rather than to close the subject. But I will go so far as to say that I believe the problems of systems engineering on supersonic aircraft are not, in any way, beyond our technical control, either in design or in maintenance. We must recognise, however, that the systems will be much more elaborate, much more costly and will probably call for larger spares holdings of components away from Base, and that realistic allowance for this, rather than a conservative allowance for utilisation, must be made in our economic studies.

Some Economic and Operational Questions

BY

C. H. JACKSON, B.Sc., A.C.G.I., D.I.C., A.F.R.Ae.S.

(Assistant Chief Engineer (S.D.), B.O.A.C.)

THE ARGUMENT for the supersonic transport seems to run as follows: it offers a new and faster way to go places; it might not cost more to do so; it ought to create new wealth, and its influence on human affairs, although not clearly foreseeable, surely will be beneficial. There is a certain amount of evidence to warrant optimism; there are issues of national prestige and well-being; the technical challenge is exciting; so "Let us somehow find finance and proceed."

Operators agree that the design and operational problems of supersonic aircraft, one of the possible great advances in transport, are complex and interesting; they might also speculate on its likely social implications, though this is an unrewarding pastime. The operators accept that the aircraft probably will be developed, for it is as much a challenge of the times as have been all the great innovations of routes and vehicles from Columbus right through to our day; but as has always been the case, before any commitment, they must seek assurance of ultimate economic soundness.

At this date designers in the U.S.A. and the U.K. have evidence that the technical problems can be overcome to a degree, and that a safe operation somewhere in the range Mach 2 to Mach 3 could be mounted with direct operating costs per seat-mile approaching those of the current subsonic jets. Such estimates will compel the major long-haul operators to study proposals and their effects on future equipment policy, even though they have not yet fully absorbed the new subsonic jets and almost all their commercial and technical development effort is being applied to these aircraft.

Clearly the subsonic jets must and will be kept in operation as long as possible in order to achieve a reliable and convenient public service with prospects of progressive fare reductions. This is not to say that operators will risk commercial or technical stagnation; but this desire to conserve and properly utilise the new equipment certainly is encouraged by economists, and others, who say that too fast a rate of technical development has been accepted.

The latter criticism is not fully agreed because every new equipment move has offered lower seat-mile costs through improved operational capability and overall design characteristics; and this record must have a powerful influence on outlooks on proposed supersonic transports.

To illustrate the point: on long-haul operations in recent years the Direct Operating Costs per Capacity Ton Mile have reduced from 18/20 pence for piston-engined aircraft, through 15 pence for the propeller-turbine aircraft, down to 12 pence for the large capacity jet transports. Beyond this there is a prospect that continued development from the current types of aircraft might reduce that last figure to less than 10 pence; some aircraft designers suggest that substantially lower costs can be achieved with new high speed subsonic aircraft employing boundary layer control.

In the overall analysis the technical and economic progress is now reflected in the low fare "economy" travel now being introduced and in the lowest fares on scheduled Atlantic operations, a touchstone for costs, which since 1950/51 have dropped about 30 per cent while retail prices have increased about 45 per cent.

In accord with this, and well before 1970, a technically feasible date for the operation of a supersonic transport, about 90 per cent of passengers will be in the lower fare bracket. Now, although operators fly in most conditions, they are unlikely to fly into the storm of opposition to an aircraft purchase that would necessitate reversing the trend by requiring more passengers who will pay more for a better service; for 10 per cent of the total would not be sufficient support for large-scale supersonic production or operation. Interest in the supersonic transport, therefore, is likely to be qualified by the view that its performance must be available at no higher, or relatively little higher, cost to the passenger.

This cost objective is the challenge to the designers' skill, and to judgment on the most suitable speed in the range Mach 2 to Mach 3; it will remain so as long as we have as our economic criterion the need to pay our way on unsubsidised operations, with adequate allocations for re-equipment; or until market researchers can prove that sufficient people will pay more, if this be necessary, to travel faster.

Of course, if the direct operating costs per seat-mile on the subsonic aircraft can be reduced substantially below present levels, then the supersonic aircraft might more readily bear a fare differential; but even so the minimum order of seat-mile costs now achieved seems likely to set the highest acceptable levels, and even this could be criticised as an unwarranted alleviation of the technical task!

Let us note what some of the designers are saying: at the moment precious little information has been released that would stand up to close economic study and some of it is disconcerting. Many papers have been presented in the U.S.A., so let me read two quotations; the first from a lecture from one of the leading U.S.A. design teams, referring to utilisation, a key economic factor, states:—

> "He (the operator) must schedule his aeroplane so that it can actually produce the capability of four jets or a dozen piston aircraft. This may mean some interchange of equipment with other airlines in order to achieve full utilisation."

Admittedly such equipment interchange is already developing between some operators; but the speaker was suggesting that the supersonic aircraft would compel the adoption of such procedures among the major operators if reasonable fare levels and economic operations are to be achieved.

The case presented probably was exaggerated because few, if any, of the published data suggest that designers are projecting first generation aircraft of more than about twice the output of the current subsonic jets.

Such outputs, coupled with predictions of traffic growth, do not on first reaction seem to warrant the fears expressed. Nevertheless, the expression of such views by the designers is an indication of the tremendously difficult design and economic balance with which they are struggling.

To take another quotation, from a U.S.A. publication: —

> " A profitable Mach 3 jet, derived by minimum adaptation of the B-70, could fly in 1967, according to Senate Preparedness sub-committee. It would carry 80 first-class or 108 tourist passengers, cruising at 1,980 m.p.h. at 70,000 ft. Operating cost would be 2·5 cents per seat-mile for domestic trans-continental and 2·75 cents trans-atlantic based on 3,000 hours annual utility."

The inference is a direct cost per seat-mile, for it is believed that direct costs were being quoted, about twice that of the current subsonic jets.

Some U.S.A. designers have suggested better results, but even in these instances, lurking behind the estimates, there is the assumption that the whole of the research and development will be carried on a military budget, or in some other way by a Government authority, and not amortised over production and borne as a normal operating cost. Such indirect subsidy might be similar in kind to the background of some current aircraft, but rather greater in degree, and equivalent perhaps to 15 per cent or more of Direct Operating Costs, dependent on design cruising speed. As far as one knows this important economic issue has not received much more than the most general consideration or expression of views.

The claims of acceptable costs often also infer the need for new economic approaches by the operator or even to suggest achievement of high load factors, by capacity or frequency restriction, so that the designer can talk of return on investment and gloss over apparently high seat-mile costs.

At this date, therefore, the operator might conclude that only on the most optimistic assumptions could designers, using standard cost methods, forecast seat-mile costs matching present levels, and that they might equally well forecast twice that level. To have shown even this possibility is a sizeable technical achievement sufficient to warrant vigorous efforts to design an acceptable aircraft; but it is still short of the forecasts that established the subsonic jet programme.

Arguments on methods of costing and economic assessments hardly seem likely to affect the position; after all, no accounting juggling can alter the cost, per seat-mile, of crewing, fuelling and maintaining the aircraft, because such costs are determined directly by the performance and design characteristics. Incidentally, referring to fuel costs, one wonders if the economic advantages claimed for the highest speeds in the Mach 2–Mach 3 range might not be somewhat reduced if kinetic heating necessitated special fuels or additives; but this is merely one of the many questions to which operators would seek answers.

Some of the indirect costs, that is, sales commission and station costs, might reduce as volume of traffic and, maybe, speed increase; but these prospects could be nullified to a degree if scheduling and preferred travelling times led to bunching of arrival and departure times. Anyway, most of such improvements could equally well apply to the subsonic operation and so only marginally better the case for the supersonic transport.

There is a possibility that, as supersonic speeds reduce inter-continental operations to something like a regional operation, some passengers might accept previously unpopular departure and arrival times for the sake of reduced fares. This might boost utilisation and annual average load factors, but the combined effects would hardly be sufficient to offset a substantial cost differential.

Engine noise in the airport environs, moreover, might become a major issue if it is allowed that the economics of supersonic transport would necessitate the selling of late-night arrivals and departures at major airports; this presumably could considerably affect decisions on the best type of engine and, therefore, the likely development and operating costs.

Annual utilisation is a key figure in cost estimates and we know that it is more difficult to schedule for high utilisation as cruise speeds increase; but maybe this would be not so much a problem as a guide, on the one hand to the cruise speed below which it is not worth struggling to go supersonic or, on the other hand, to the highest speed that would pay off having regard to the problem of scheduling for saleable travelling hours.

On balance it does not seem that the technical difficulties of planning for high utilisation would be eased by equipment pooling between airlines. Of course, estimation of acceptable economic results would be easier if pooling is postulated as a means of avoiding the offering of surplus capacity by competing airlines. Maybe this is what the U.S.A. speaker mentioned earlier was really suggesting; but the resultant reduction of the number of aircraft and spares would cause some penalty on costs and anyway is hardly likely to encourage manufacturers even if it could be one of the predictable results of supersonic transport. Moreover, even if such assumptions are made it does not follow that the convenience of passengers would be well served if the consequence were to be a lower frequency of services on some routes, or a reduced choice of carrier, in return for the reduced journey times.

There seems little in the likely engineering and

operational characteristics of supersonic transport that could alleviate any cost problems; if anything the reverse would apply. The engineering systems of a supersonic transport probably will not be all that new or disturbing, but in some instances they might necessitate a higher standard of maintenance skill, and therefore higher engineering costs than at present. A more important matter would be the likely greater dependence of operational reliability on system reliability, because the supersonic transport probably could not afford the percentage of delayed departures due to engineering irregularities from which we now suffer; for instance, on a high density route even a 10-minute delay might mean losing the slot in Air Traffic Control's departure schedule. System reliability, therefore, could be basic to operational economics and calls for major efforts from aircraft and component designers.

The supersonic "bang" in cruise flight might raise an even more critical issue, because designers' cost estimates seem to assume subsonic flight only to and from about 35,000 ft. But what happens to even the present economic forecasts if the bang en route necessitates subsonic flight over Europe and large parts of the U.S.A., and expensive coastal diversions around India and South America etc.? The illustrations show what might result from such a situation; although such operational prospects are not altogether discouraging, their implications for costs suggest the need of a careful assessment of Variable Geometry projects as a cure for uneconomic subsonic flight.

If the sonic bang restricted the aircraft to transoceanic operations, primarily the Atlantic, it could be suggested that the development of large capacity aircraft with correspondingly lower seat-mile costs, if such a trend indeed applies to supersonic projects, would be warranted; this conclusion would be regrettable, since the Atlantic route could not show the speed of the supersonic transport to best advantage, particularly if little is done to shorten journey times between airport and city-centre.

It is clear, therefore, that a decision on the existence or otherwise of the en-route supersonic bang as a sociological problem is vital to the economics of supersonic operations and to decisions on the leading characteristics and commercial application of any project.

Outside the field of airworthiness a first and superficial reaction might be that the general operational problems of a supersonic transport will be only an amplification of those encountered with the wide-scale introduction of subsonic jet transports, particularly on regional operations. Some of these problems, for example the need or otherwise for automatic communication and flight control, for the proper integration of the flight and all the ground support, might be resolved in advance for the regional operators in Europe and the U.S.A. The systems developed might easily cater for the shrinking of a hemisphere of trunk routes to the time-tables of a European operation; one might even hope that by then the Customs and Immigration procedures will have been correspondingly reduced.

But the operators know that they cannot just make general operational assumptions and wait. The time has come for them to ask detailed operational questions affecting the economic practicability of projects, almost as significantly as the over-riding question of the supersonic bang; for example, it is not known yet if autolanding will be available, or necessary, because of aircraft flight handling characteristics and the need to maintain present standards of schedule regularity. Alternatively, will scheduling and economics force the need for higher than present standards of regularity, and will this necessitate auto-landing, even if aircraft handling characteristics are much the same as at present?

Operators might consider the prospect of autolanding allowing reduction of fuel reserves directly to better the design efficiency and economic results; but they must also ask if such reserves anyway would be decided by emergency conditions such as subsonic enroute diversion after engine failure? As they think round such questions they must also ask if it would be better *not* to rely on the additional electronic aids and automation, and instead put their money and effort into later Variable Geometry proposals which, it is claimed, would avoid such critically related economic and operational problems?

Will Air Traffic Control be able to cope with the separation problem, with high frequency operation of moderate capacity aircraft? Is this problem likely to be worse in cruise flight, or on climb and descent below 35,000 ft., where both supersonic and subsonic aircraft might be seeking their best, and conflicting flight profiles?

Are en-route temperature changes likely to be a problem, because of systems on Mach 3 aircraft, or because of primary materials anywhere in the cruise speed range of Mach 2 to Mach 3? If so, what operational equipment and techniques will be needed to detect and avoid, or to reduce speed before entry, regions of high temperature or, for that matter, regions of turbulence or hail?

Will scheduling problems be manageable if high first cost and/or some problems of utilisation necessitate a degree of equipment pooling between operators? In that event would national differences on cockpit requirements then be resolved, or would equipment pooling almost mean crew pooling, and in the end virtually the merging of certain airlines, at least in respect of supersonic operations?

In the present state of knowledge the list of operators' technical questions is endless. No doubt as they are recognised ways of resolving them will be found; but, like Columbus, designers and operators will still face the question: "How can economic justification be demonstrated?" If the designers can meet the operators' present cost objectives the questions disappear. If they fall short of it, but not sufficiently to warrant outright rejection, could operators change their economic thinking or their competitive relationships; or could they hope to support it on the returns from the mass travel market they are creating with subsonic aircraft? Or will economists and market researchers by

then have proved the existence of a large enough market for higher fares?

Probably no operator in the world could answer such questions. Most likely in the U.K., U.S.A., or elsewhere, the operators must wait a year or more for the minimum information for even provisional judgment; meanwhile they must re-examine the likely trends with their subsonic aircraft and consider any changes of present outlook on tariffs and economics that might influence reactions to the forthcoming projects.

Summary of Discussion

BY

MAUREEN E. MICHAEL, M.A., Grad.R.Ae.S.

Part I.

Discussion after Papers by Wilde, James and Megson

In opening the discussion on Mr. Wilde's paper, a speaker from B.O.A.C. asked what the 12° angle of incidence would mean in terms of what the pilot sees and whether his downward view of the airport would be affected. The reply indicated that on the landing approach the fuselage would probably be inclined at 10° to the horizontal and that in order to give the pilot a reasonable view some sort of variable geometry of the nose would be necessary.

When asked what he thought was a reasonable view from the pilot's point of view, Mr. Wilde said that various standards had been laid down but opinions varied about these. Because of the slender configuration of the nose there would be a good view at the sides, and a certain minimum standard could be achieved for the direct front view. It was too early to say that this could be made as good as in the best of the present subsonic aircraft but he was confident that it could be made acceptable to the operator. Mock-ups were being prepared.

Asked why the landing speed of the aircraft came down with reduction in wing area, Mr. Wilde replied that on landing the wing loading would increase as the area decreased, but this would be more than balanced by the associated increase in lift-curve slope. Mr. Wilde amplified this statement by showing that the lift-curve slope of relatively slender wings was made up of two parts, a linear part varying directly with aspect ratio and an additional non-linear term which was a function of α above no lift. His diagram showed a lift-curve slope derived by Küchemann which seemed to fit the experimental points well.

Concerning a variable geometry wing, Mr. Wilde thought that this would probably be essential once the speed was greater than Mach 2. Mr. M. B. Morgan, in his lecture to the Society in March 1960, had indicated that a Mach 2 aeroplane would need good low speed characteristics, and there was no doubt that variable geometry would improve these, but designers would probably prefer not to make the design of the first generation supersonic transports too complicated.

Mr. Wilde was then asked what the effect on the transonic acceleration would be with the compromise aeroplane he had suggested and he replied that the acceleration phase would improve as the wing area was reduced, very largely because of the increasing engine thrust, the thrust required being a direct function of C_{D_0} which is increasing. Asked if reduction in wing area would mean that the aircraft would cruise at a lower height, he said that this was not so; the variation in cruising altitude for the aircraft he had considered was only about 2,000 ft.

A speaker from Hunting Aircraft Ltd. asked about the cost of such an optimum aeroplane; although the wing area was small the engines would cost more per pound of aeroplane and surely this would increase the total cost?

Mr. Wilde said that this had appeared to be the case but in checking up on this, using current quotations for engines, he had found that in the example he quoted in his paper the direct operating costs per seat mile varied by less than two per cent. This was taking a fixed cost per pound of airframe and a fixed cost per pound for the engines. A speaker then remarked that in making the compromise there was a possibility of "getting some weight back" by employing favourable interference between the separating flow fields of the wing bodies. He would like to hear views on this and in particular ideas as to how far back along the root one would be able to go by this method.

Mr. Wilde pointed out that where the wing and the fuselage were distinguishable features of the layout with fuselage depth of about 10 ft. and wing maximum depth of about 5½ ft., the design could not be regarded as integrated and there would therefore be different fields of flow on the wing and fuselage. For years now people had been attempting to make these interfere in a way favourable to the lift-drag ratio, but he could not divulge how they were intending to set about this. An example he quoted indicated the order of improvement which might be obtained and also highlighted the need for very accurate measurements to be made in the wind tunnel.

Opening the discussion on Mr. James's paper, a speaker from the Air Registration Board remarked that the argument that a safety factor was already built into existing aircraft that took care of the heating effects was ingenious, but he wondered if it were true. Heating effects on existing aircraft were confined to relatively small areas and he was uncertain whether sufficient experience had been gained in dealing with heating effects over complete structures as would be the case in a supersonic air liner. Mr. James pointed out that in existing cases of integral fuel tanks the safety factor had to cover the whole of the primary wing structure.

When asked if it was as easy to calculate the thermal stresses in a structure as it was to calculate stresses in a cold structure, Mr. James replied that this must be looked at in its proper context. The stresses were calculated from loads derived from aerodynamic data which were probably more unreliable in the high subsonic range than in the supersonic range. The safety factor could not be taken as dealing with the purely structural aspects. Another speaker from the A.R.B. thought that in order to account for the thermal effects they should look again at the safety factor concept. The present concept of laying down the proof requirement and a safety factor of 1·5 was laid down twenty or thirty years ago and he wondered how relevant it was today. At the moment the limit load was regarded as something that would occur to every aeroplane in the fleet once in its lifetime. It was now necessary to cater for the one load that one aircraft would experience that was higher than the limit load; once this was accepted the safety factor could probably be brought down to 1·2 and it would be possible to account properly for thermal strains.

Mr. James thought the difficulty in such an approach would be in knowing just how to break down the safety factor in order to build it up again to cover the new effects. At present the 1·5 factor covered errors of failing stress, load distribution and so on and errors in estimation of aerodynamic loads. It would be difficult to break it down into elements covering each variable, some of which, *e.g.* temperature effects, had been covered unwittingly.

Referring to one of Mr. James's diagrams on creep life, a questioner asked how the figure of 30,000 hours was arrived at; he thought 30,000 hours utilisation for a supersonic air liner was more than could be expected in a ten year life and that the figure should be halved. He also wondered if it could be assumed that the thermal effect was cumulative over the whole life of the aeroplane.

Mr. James admitted that the figure of 30,000 hours had been taken out of the hat and was probably too high, but it had to be recognised that an aircraft originally designed to last for ten years would in all probability be required to last three times as long as that. He had chosen a long period to show that creep, as affecting direct permanent deformation, and being related to the proof strain requirement, was not likely to be critical even over such a long period. Also he wished to draw attention to the need to consider apparently secondary effects arising out of the small amount of creep which would occur.

A pilot from B.O.A.C. then referred to Mr. James's statement that the supersonic aeroplane would be temperature placarded; in view of the obvious difficulties in estimating temperatures at altitude, how restrictive would the temperature placarding be? At supersonic speeds the pilot would have a very big problem in slowing down the aircraft if there were a sudden temperature rise of 10° or 15°C as might occur. Mr. James replied that he could not give a definite answer; he was concerned with the variation in the structural temperatures and assumed that at the point chosen to indicate the temperature placard limitation there would have to be temperature control within small limits.

He was then asked if he would like to see structural testing done on these aircraft and in particular what sort of factor would be applied during, say, proof testing. To this he replied that he did not see clearly how tests at this level would be done. Under the current system proof loads were often held for several hours. A creep curve would show that a two hour exposure to proof load could result in deformation quite different to that existing immediately after the initial application of the load. The difficulties would only be exaggerated by putting a factor on the temperature.

When asked about the possibility of using beryllium as a structural material for supersonic transports, Dr. Megson agreed that it would be a very attractive material if it could be used, because of its low density, but so far all attempts to alloy it had resulted in very brittle products and metallurgically speaking it appeared that serious difficulties would need to be overcome. He was also asked about the possible uses of magnesium and molybdenum. Concerning these he agreed that magnesium would certainly have its place for components inside the aircraft, but not for the outside structure. Molybdenum would be awkward to use because of its tendency to oxidise, but efforts were being made to protect it for use in engine parts and so on. To a speaker who asked if an aircraft made of titanium were feasible, he replied that technically it appeared possible, although the high cost was a serious disadvantage.

Another speaker asked for Dr. Megson's views on a suitable non-inflammable fluid for hydraulic systems at high temperatures. The most promising material here seemed to be a modified silicone of American origin; this had proved more satisfactory than the ordinary hydrocarbon or phosphate type of fluid. It was adequately non-inflammable and had reasonable lubrication characteristics, but there was room for improvement. Returning to the question of elastomers and sealants it was noted that temperatures had been quoted for lives of the order of thirty hours, a speaker enquired what the temperature limitations were for a life of 10,000 or 30,000 hours.

Dr. Megson thought there might be a temperature for which a life of 10,000 hours for Viton A could be quoted but could not say without further reference. He did not know what its flexibility would be at sub-zero temperatures.

Part II.

Discussion after Papers by Pearson, Reed, Sutcliffe and Jameson

The two papers on propulsion and the second aerodynamics paper were given as a group and a joint discussion on all three papers was held. It was opened by a speaker from Vickers-Armstrongs (Aircraft) Ltd., who noted that two of the papers had made quite strong arguments in favour of a variable geometry aircraft. Such an aircraft would achieve better economy if it had to cruise subsonically over populated territory.

Mr. Sutcliffe pointed out that although variable geometry would help by flying the subsonic portion more economically and easing the thrust needed for take-off, it was not the extra fuel that would otherwise be needed to fly subsonically that put the operating cost up, but the reduction in block speed.

A speaker from R.A.E., Farnborough, said that he would like to emphasise the importance of the intake problem; the flow in supersonic air intakes was very complicated and the difficulty of obtaining an adequate pressure rise with a low drag penalty in the intake was much greater than that associated with achieving an adequate wing lift-drag ratio. In the pods *versus* buried installation argument he stated that he was a proponent of integration. Thinking back over the past twenty years or so there had been some very successful British designs with buried engines and he thought these should not be changed without giving serious thought to the further sophistication of integrated shapes; he was not suggesting that sentiment or aesthetics should dictate the design of an aircraft but he understood that on the technical side the drag comparisons were fairly evenly balanced. Thought had to be given to the treatment of the boundary layer; so far no research had been done which would show how to incorporate the pre-intake compression into the aircraft contour. A contributor from N.G.T.E. thought that one aspect which ought to be mentioned in favour of pods was the ease of testing a complete power plant. If the engine designers were faced with the prospect of developing an engine in a box with two others all sharing a common intake, the problem of development testing would be very difficult.

An airline pilot then asked Mr. Reed about the table he had shown in which a figure was deduced for the noise value of an aircraft at the 4·2 mile point. He wondered on what type of procedure this was based. Mr. Reed replied that a throttling technique was assumed which provided a certain margin of climb when one engine was out; he thought this was similar to that used by one of the airlines operating from London Airport.

A speaker from the R.A.E. remarked that much attention had been given to the sonic boom problem and to noise in the vicinity of airports, but he thought that mention should be made of engine noise during take-off and climb as a fatigue producing agency; this was already quite a problem with subsonic jets and no doubt it would be a bigger problem with supersonic aircraft. He also mentioned the fatigue effect of boundary layer noise because recent measurements had shown that a level of 140 decibels might be expected in cruise and this was capable of producing fatigue.

Mr. Pearson did not think the panel could comment on the question of boundary layer noise but on the general fatigue from engine noise, work done some years ago had shown that rear mounting of engines was better than the type of installation used in the Boeing 707 and the DC-8

The supersonic transport would achieve a certain level of safety with engines mounted at the rear but he would not say that there would not be any fatigue failures.

A speaker from Handley Page Ltd. said that his firm had done some calculations relating to boundary layer noise based on some data from Douglas Aircraft Ltd. and it seemed that it would be no worse than that now being experienced on the V-bombers.

Returning to the pods *versus* integration argument, a speaker from Hawker Siddeley Aviation Ltd. came out strongly in favour of pods. If one intake failed on an integrated installation it could lead to perhaps a triple failure of the engines and from a safety point of view this could not be tolerated. If it could not be proved that an integrated design gave better drag than a podded one then there was no case for integral engine installation.

Mr. Pearson would not accept these statements without further proof and he thought that a variable circular intake would be a difficult practical proposition compared with the two-dimensional one.

On the question of the sonic boom, the same speaker from Hawker Siddeley pointed out that on the basis of existing evidence, the boom from a supersonic transport weighing 200,000 lb. would be termed objectionable. It seemed likely therefore that the Governments of such densely populated areas as the United States and the United Kingdom would introduce legislation prohibiting supersonic flights over land and the Governments of more sparsely populated areas must be expected to follow suit. This would have an obvious effect on airline economics. Fig. A showed a comparison of the cost of operating a supersonic transport under these conditions over various routes compared with one operating unrestrictedly.

The datum for costs in this figure was that based on the London-New York run where the sonic boom had little or no effect. Various sectors on current routes had been taken and the operating costs evaluated if the whole flight were flown without restrictions. These were represented by the circle at the bottom of the vertical lines on Fig. A. The numbers alongside these circles were the journey times on the particular route. The effect of restrictions due to sonic boom on operating costs was shown by the triangle on the top of the vertical lines and the numbers there referred to the journey times, assuming over-land flight at $M=0{\cdot}9$.

Another speaker from Hawker Siddeley continued the theme of how noise would affect the pattern of operation of supersonic transports. He felt that this pattern would be determined by the traffic intensity and the potential for traffic growth on the routes considered. His first diagram (Fig. B) showed the possible pattern of air traffic flow in

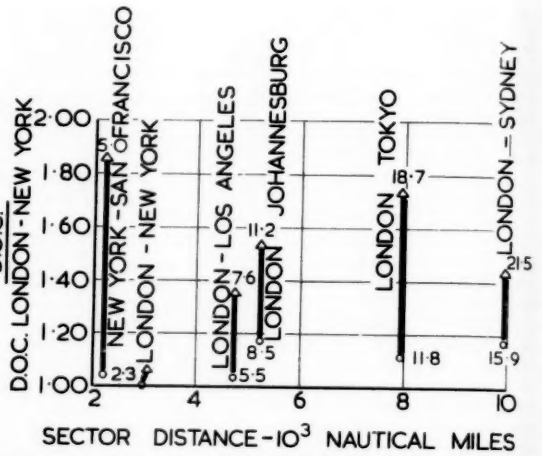

FIGURE A. Effect of speed restrictions on direct operating costs of a $M=2{\cdot}2$ transport.

the early 1970s, and the proportion of the total supersonic traffic that could be expected on each of the six main streams. If supersonic flights were prohibited over land this picture would be drastically altered. The overall effect of world geography and the positioning of the major industrial cities was likely to influence very drastically the pattern of supersonic transport aircraft operation. He summarised the total effect on large supersonic transport aircraft markets in 1970, assuming rates of growth for passenger traffic comparable with recent forecasts, taking as the yardstick a 100-seat Mach 2·2 air liner producing 300 million seat-miles per annum. For B.O.A.C. and the Commonwealth air carriers, although there was a theoretical requirement for 24 aircraft, only 59 per cent of these could be used because of restrictions on over-land operation. For the United States overseas airlines, although 31 aircraft might be used in theory, only 49 per cent of these could in fact be operated. For the U.S. domestic airlines an unrestricted total of 40 aircraft would be reduced to 16 per cent of this, probably only six or seven aircraft. For other world air carriers although 34 aircraft were the theoretical requirement, only 62 per cent would probably be required. Thus for a total of 130 to 140 large supersonic aircraft a residue of only 60 aircraft was likely to be operational on the major world routes in the 1970s (Fig. C).

Opening the discussion on Mr. Jameson's paper, a

FIGURE B. Unrestricted Supersonic Air Transport in 1970. The figures indicate the percentage of total traffic on the main world air routes.

Area	*Percentage of total required if boom restrictive*
British Commonwealth including B.O.A.C.	59%
U.S. International	49%
U.S. Domestic	16%
Other world carriers	62%
Total	43%

FIGURE C. Effect of sonic boom on world market.

speaker from Hawker Siddeley asked about icing. Although this would not be a problem during supersonic flight and the early stages of reducing speed, the supersonic transport aircraft had to be capable of the same operation as a subsonic aircraft and it would have to fly frequently in subsonic conditions where ice build-up would occur. With very long leading edges and large areas ahead of the intakes a considerable amount of heat would be needed to safeguard the engines; he wondered if there would be any relaxation in the requirements here.

Mr. Jameson thought this would not be so; the flight plan might require less endurance but the general spectrum of ice which the aircraft had to be able to resist would be the same. He was then asked about cracks in the windscreen which could result in a pressure failure and possibly disaster. A supersonic aircraft was less able to slow down to avoid hail than a subsonic one and the transparencies would have to work over a greater range of environmental conditions. Was it considered that present methods of testing were adequate for proving this sort of structure?

Mr. Jameson agreed that there was a need to develop a really reliable window material; the best hope at the moment lay in using enough redundant structure to ensure that a crack did not produce disaster. Testing of pressure cabins was now at an interesting stage and it could well be that improvements would have to be made in design.

Mr. Jameson was asked for his views on the automation of systems to relieve the flight engineer of his onerous task, leaving him free to overlook the instruments. Could automatic devices be regarded as a substitute for a crew member? Here Mr. Jameson thought it was necessary to proceed with care, for although a good case could be made for automation, it would be easy for the crew to get to the state of worrying whether the automatic devices were working. There would probably have to be a crew member who was especially responsible for these devices and who would not be directly interchangeable with the pilot. When asked about the possibility of coupling the power plants to make things easier for the pilot to handle, he thought it was necessary to avoid compromising the independence of the engines in any way: if they were coupled the probability of the coupling failing in a dangerous way would have to be extremely remote. The case could only be considered in terms of say, 20 engines, which could be grouped together in fives and regarded as four power units.

The next speaker said he would like to emphasise the warning already given against trying to cover too many things when considering the supersonic transport. They must consider what was really new in this respect. A case in point was the hail aspect; perhaps a virtue could be made of the sonic boom effect by saying that because of it the aeroplane would not go supersonic until it was above the hail spectrum and therefore the hail problem would be no worse than for present day aircraft.

In reply, Mr. Jameson said that a lot more information was still needed concerning hail, and operators had been fortunate in getting off lightly so far. At present when hail damage was caused at low speeds the pilot was usually aware of it and had no cause to go faster afterwards; the danger with supersonic aircraft would be that the pilot might not be aware of hail damage caused when he was flying subsonically and would then try to fly supersonically with a damaged aircraft.

PART III.

DISCUSSION AFTER PAPERS BY ORMONROYD, MORGAN AND JACKSON

The Chairman, Sir George Gardner, opened the discussion on Captain Ormonroyd's paper by remarking that from his own experience the paravisual director made it easy for someone who was not a pilot to fly the aircraft.

A B.O.A.C. pilot then asked for Captain Ormonroyd's views on the approach and landing of the supersonic transport. The pilot would be a long way in front of the wheels and inclined upwards at an angle of 10° or 15°; despite the optimism expressed earlier in providing the pilot with a 25° downward view, he frankly did not believe this would be so.

Captain Ormonroyd replied that he was a great believer in automatic devices and thought that this was how the problem should be tackled, even if four auto-pilots were necessary to ensure safety. The questioner agreed that this was perhaps the right idea but wondered how many airports even in 1970 would be equipped to handle fully automatic landings. He felt that provision must be made for the pilot to land the aircraft manually. Even so, Captain Ormonroyd stressed that the pilot must have some automatic assistance, otherwise the weather minima for manual landings would be so high that regularity would be bound to suffer.

Mr. Jameson asked why, if it was so desirable to get the orthodox flying controls out of the way in order to see the instruments, this had not been done before. It would be expensive in proving time on a supersonic aircraft if it had not already been proved on a subsonic one.

Captain Ormonroyd thought that some experience had already been gained on these lines with the Vulcan. With no cables or large forces involved it should not be difficult to move the controls to one side and put them into the form of a tiller.

A speaker from the R.A.E. pointed out that any automatic element was liable to go wrong either by failing when called upon to perform or by performing inadvertently; this implied that a good case could be made for improving monitoring and display, thus allowing the crew to take rational and intelligent action.

Captain Ormonroyd agreed and quoted the case of the Trident where B.E.A. planned to use the paravisual director as a monitor for the duplex auto-pilot. To guard against the pilot disconnecting a serviceable auto-pilot and acting on the demand of the director in the event of a malfunction of the director itself, however, it would be necessary to blank off the P.V.D. unit during the automatic approach and landing procedure and arrange for the shutter to be released if and when the auto-pilot failed.

Opening the discussion on Mr. Morgan's paper (which had been read for him by Mr. W. B. Shaw), a speaker from Hawker Siddeley commented that systems for supersonic transport aircraft would be more complex than hitherto and would need more components. Higher standards would be needed for things like air conditioning and this would mean closer tolerances on the components. All this seemed to add up to the fact that a higher standard of detailed engineering would be necessary.

Mr. Shaw replied that this was not so; the need for greater reliability applied to both subsonic and supersonic aircraft and the standard of engineering required on supersonic aeroplanes would be the same as that for the subsonic jets. The second point that the speaker made concerned multiplication of systems. He admitted that to obtain higher component reliability would cost more initially but it would be of great benefit to future aircraft if it reduced the number of channels needed. No increase in the number of channels could be tolerated, therefore the reliability of components must be improved so that multiplication could be reduced. Mr. Shaw agreed but saw no prospect of achieving the necessary standard for safety with single channel equipment in the forseeable future.

This led a speaker from the R.A.E. to state that there

was apparently some conflict between flight safety and ground serviceability, and since for commercial aircraft ground serviceability was of paramount interest economically the question came to the fore.

Mr. Shaw agreed that this was so and he would therefore oppose multiplication of equipment except where it was needed for safety, but where safety was concerned there were too many and too varied causes of failure to allow for a successful attack on them all at once that might lead to a simple system.

Asked by the Chairman why he did not favour automatic check-out equipment before flight, Mr. Shaw replied that the difficulty here was in establishing the lowest common multiple of a large number of tolerances which would allow the automatic check-out system to pass out the systems as serviceable without failing them every time. A manual check established that the systems functioned whereas an automatic check would check all components, outputs and so on. Because in most cases the aircraft had just landed from a successful flight and a certain degree of safety was given by multiplication of systems it would not be necessary to check everything in complete detail every time. However, where a faulty system was known to exist test equipment should certainly be used to locate the faulty component.

A speaker from Vickers-Armstrongs (Aircraft) Ltd. opened the discussion on Mr. Jackson's paper by asking if the operators would be interested if they were offered the same seat-mile cost on the 1,500 miles range as had been offered for the 3,000 mile range, *i.e.* about 1½d.

Mr. Jackson replied that they would obviously be interested in having a low cost for such a stage length but they had to recognise that a large proportion of their traffic was trans-Atlantic. They could not afford to get excited about the low cost on a 1,500 mile stage unless they knew what was involved on the longer stages. Asked if he was worried that 1970 might see a very unbalanced transport system with New York nearer than Rome and Australia than Cairo, he said this was a real possibility. Flight control systems would have to be developed into which the supersonic transport aircraft could be fitted.

A speaker from Handley Page Ltd. noted that Mr. Jackson had referred to the benefits of laminar flow as applied to subsonic aircraft economics. His firm had been doing some work on the application of boundary layer control not only on subsonic aeroplanes but on supersonic ones as well. Preliminary results were compared to a standard of 12 pence per ton mile for a current subsonic jet as follows:—

	Pence/ton mile
Current subsonic jet	12
Large subsonic laminar flow aeroplane	6
Development of current subsonic jets	9-10
Conventional Mach 2·0 aeroplane ..	15
Mach 2·0 aeroplane with suction wing	11
Mach 2·0 areoplane with slewed wing	10

He thought even better results might be obtained by slewing a suction wing.

An airline pilot pointed out that the introduction of supersonic transport aircraft would bring large Air Traffic Control problems. Although jets were first introduced into service more than ten years ago A.T.C. problems still resulted in them being operated in a most uneconomical manner. He could not see A.T.C. authorities throughout the world being ready to accept supersonic aircraft in 1970. Other problems were take-off and landing both of which he thought would be difficult, navigation which would have to be fully automatic, weather—from past experience he did not believe those who told him there would be no weather problem at the cruising altitude of these aircraft—and transition from subsonic to supersonic flight. He thought it would be difficult to find the necessary large area of clear air at 30,000 ft. each time.

A speaker from the Ministry of Aviation noted that Mr. Jackson seemed somewhat pessimistic about supersonic transport aircraft; had the operators felt the same way twenty years ago about the propect of operating the high subsonic jets? Mr. Jackson replied that at the same stage in the high subsonic jet development they had seen rather better economic prospects than they now saw for the supersonic aircraft. This did not mean that they were not enthusiastic about the supersonic projects, they were, and they regarded them as inevitable, but the costs must be brought down to a reasonable level. Asked if he really thought the passengers wanted to fly supersonically, he said his impression was that people did want shorter journey times provided the price was right. It was suggested that the money being spent on this project could perhaps be better used trying to cut down the time spent travelling between city centre and airport, but he thought that this would prove even more expensive than developing a supersonic aircraft.

Mr. Sutcliffe asked if it would be possible to explore the effect of the indirect operating costs on the economics; so far everything seemed to be related to direct operating costs. If the indirect costs for a supersonic aeroplane were the same per hour, not the same per mile, as those for a subsonic one, then the indirect costs would be roughly halved and this might outweigh the increase in direct costs.

Mr. Jackson agreed that the indirect costs must be considered but for the first presentation of proposals and indication of trends it was necessary to rely on the direct costs of operating the aircraft and these varied over a very wide range, particularly if any sort of allowance was made for periods of subsonic operation.

Part IV.

General Discussion

Sir George Gardner invited Dr. A. E. Russell, of Bristol Aircraft Ltd., to open the general discussion by giving a general picture of the problems surrounding the development of supersonic transport aircraft as he saw them.

Dr. Russell said that the justification for a supersonic transport must be based on its potential economics. The conditions were that operating costs, when reviewed in competition with alternative forms of air transportation, could attract a sufficient volume of traffic when operated on typical world-wide airline networks.

Four primary factors were involved in an overall analysis, all of which might have to suffer adverse trends imposed by particular important side issues. All of these primary factors were in some way inter-related and none could be given simple numerical values.

Some arbitrary recognisable split was necessary to bring cohesion into a general discussion.

The first factor was a compound product of speed and lift-drag ratio. Evaluation in supersonic cruise alone could be misleading for the subsonic and transition phases had a bearing on the optimum result. Progressively increasing cruising speed yielded proportionately less gain in block speed, also lift drag ratios after a sharp decline beyond a Mach number of one continued thereafter in a downward direction. Moreover cruise conditions controlled only about 50 per cent of the total fuel carried. So while speeds substantially above $M=1$ were necessary to compensate for comparatively low lift-drag ratios, pressing this trend continuously lead to diminishing returns. For examp'e, it was apparent that the improvement in the compounded function containing L/D, block speed and fuel factors resulting from raising the cruise from $M=2{\cdot}2$ to $M=3$, would not be much greater than 10 per cent, as compared with the corresponding speed rise of 36 per cent.

Temperature effects on materials intruded above $M=2{\cdot}2$ and thereby introduced a discontinuity, hence they were led to focus attention on achieving the best possible lift-drag ratio at this speed while maintaining satisfactory subsonic characteristics. This was the predominant problem in the design of a supersonic air liner.

He suggested that the choice of engine should be taken

as the next important factor, not because it presented undue technical perplexity but because it was essential that the performance estimates of the engine designers should be achieved. The fuel load which must be carried was an unusually high percentage of the take-off weight so an unexpected rise in specific fuel consumption would have serious consequences. The technical horizons of engine designers were more clearly visible than those of the aircraft designer and it was therefore easier for them to define their targets. Not before $M=3$ was approached did radical departure from present practice need consideration.

In the $M=2$ régime only narrow technical margins separated turbo-jets from turbo-fans. These tended to offset each other so that overall equality of aircraft performance was obtained with either form of engine.

On the subjective issue of noise, the turbo-fan had an advantage, although the turbo-jet could not be rejected for its predicted level fell inside present tolerated limits.

The third factor of primary importance was the reliability of detail weight estimates. In the first place there were many essentially novel features in both the airframe and in the systems so that previous records and statistics were not always appropriate. Also, the consequences of unreliability of equipment in service would lead to exceptional embarrassment in traffic handling and scheduling. The temptation to pile on multiplexed systems would be great and many technical decisions must be taken without precedent for support.

In the second place, the helix of the weight performance spiral was unusually steep. It was obvious that general detailed analysis must be taken to an advanced stage before a comprehensive and a reliable weight statement could be produced. The inter-relationships between aerodynamic, engine and weight data demanded extensive exploration before the final version of a supersonic transport could be established.

Turning to the fourth and controlling factor in any form of civil transport, namely operating costs; normally costs were reduced to an hourly rate, so it was interesting to compare passenger-miles per hour per lb. of A.P.S. weight, *i.e.* potential hourly revenue earning capacity per unit weight. Using figures obtained from reliable sources (two from the United States), the relative numbers were 0·65, 1·0 and 1·0 for the Boeing 707, a $M=2{\cdot}2$ and a $M=3$ transport respectively. It should be noted that both supersonic aircraft were equal in this respect and 50 per cent better than the 707.

Using the subsonic air liner as a base for fuel costs per passenger mile the $M=2{\cdot}2$ and the $M=3$ aircraft were 50 per cent and 100 per cent higher respectively. Most of the remaining constituents of direct operating costs were

Aircraft A	Boeing 707		174 passengers	
Aircraft B	M 2·2 air liner		130 passengers	
Aircraft C*	M 3 air liner		100 passengers	
	*Ref. N.A.S.A. Tech. Note D.423			
	A	B	C	
A.P.S. weight	135,000	150,000	150,000	
Passengers	174	130	100	
Block speed m.p.h.	505	1,150	1,480	
Passenger m.p.h./A.P.S.	0·65	1·0	1·0	—(1)
Fuel costs/passenger mile	0·28d	0·45d	0·62d	
Total D.O.C.	1·15d	1·15d	1·15d	
Difference	0·87d	0·70d	0·53d	—(2)
(1) × (2)	0·56	0·70	0·53	
Ratio cost/lb. A.P.S.	1·0	1·25	0·95	
		+10%		
Total D.O.C.	1·15	1·27	1·27	
Difference	0·87	0·82	0·65	—(3)
(1) × (3)	0·56	0·82	0·65	
Ratio cost/lb. A.P.S.	1·0	1·45	1·16	

FIGURE D. Comparative costs

directly proportional to aircraft cost so for equal seat mile costs for the three aircraft, by differences from fuel costs, it was possible to infer that the average cost of the aircraft per lb. weight must be of the order of 1:1·2:0·95. If a 10 per cent higher direct operating cost were allowed for the supersonic aircraft the ratios became 1:1·4:1·16. This comparison was based on Boeing 707 with 174 seats.

There was no reason why, under similar cost of living conditions, a supersonic air liner constructed in light alloy should be more difficult or more expensive to manufacture than a conventional high aspect ratio subsonic aircraft of equal weight. But the numbers produced must be large enough so that the share of development costs in the total selling price was not greater than about 20 per cent.

On the same argument the basic cost of a steel or titanium aircraft started at an impossible disadvantage for the selling cost per lb. must be no greater than existing aircraft. Sandwich plate was currently quoted in the United States at $200 per lb. against $25 per lb. for conventional light alloy. There was no need to drag in a not inconsiderable rise in development cost.

This argument indicated that a $M=2{\cdot}2$ air liner made sense and a $M=3$ air liner nonsense.

The first speaker after Dr. Russell did not agree with him that there was a discrepancy between his conclusion on the sonic boom and the figure shown by Mr. Sutcliffe. For normal types of aeroplane the sonic boom could be divided into two distinct parts; at low altitude intensity of the boom was dictated by the cross section of the aeroplane and at high altitude it was dictated mainly by the weight. If the aircraft was light then over most of the altitude range the boom intensity was determined by the cross-sectional area and the rate of change of sonic boom intensity with altitude was quite large. For a heavy aircraft at high altitude the boom intensity was determined by weight and in this case the rate of change of intensity of the sonic boom with altitude was small. He maintained that the NASA theoretical calculations quoted by Dr. Russell (Fig. D) were wrong at high altitudes, being based on the old theory which assumed the effective lift was zero and only took the volume of the aircraft into account.

Dr. Russell pointed out that he was not quoting theory. NASA flight studies had shown that increasing altitude was a very powerful factor in reducing sonic boom intensity. The speaker said that the NASA tests had only been done with relatively light aircraft where, in any case, the full theory would show the lift effects to be very small.

A speaker from the R.A.E., while not disputing the high pressure levels that would be obtained, felt that some of the gloom should be dispelled. He thought there was a real and an imaginary side to the problem and the difficulty would be in isolating these when trying to assess the acceptable levels. People objected to things if they were avoidable and if there was someone to complain to, but he noticed that loud thunder came into the same pressure band as the supersonic boom and he did not think people complained about thunder. People would gradually learn to accept the noise; tests recently made on rats subjected to loud bangs showed that though disturbed at first they gradually became accustomed to them.

Another speaker from the R.A.E. returned to the materials field. It had been implied that it was speed that restricted the use of aluminium alloys above Mach 2·0, but it was of course the increase in temperature that imposed a limit. He wondered if the extra weight needed for cooling equipment enabling aluminium to be used for a Mach 3·0 aircraft would be less than the extra weight due to the high density of steel.

Dr. Russell thought it would be possible to make a stainless steel sandwich structure for a $M=3$ aircraft that would be lighter than an aluminium one. Nobody had claimed to be able to work aluminium alloys above 130°C and he felt for the main structure aluminium would have to be limited to a $M=2{\cdot}2$ or possibly 2·5 aircraft.

A speaker from Vickers-Armstrongs (Aircraft) Ltd. noted that it seemed to be assumed that the variability of

parameters (*e.g.* in performance) would be the same for supersonic aircraft as for the subsonic ones, but was this true? He thought it would be a long time before there was sufficient information on supersonic aircraft to settle this point. A better approach would be to take positive action now to reduce the variability of such parameters that were known to be sensitive. He agreed with Captain Ormonroyd that there should be a take-off director and he thought a complete automatic take-off should be considered. This would reduce errors in speed during climb. He would like to know if Captain Ormonroyd thought that it would be possible, in the event of a failure, for the pilot to take over a landing when only 50 ft. above the ground. Captain Ormonroyd could not answer this without knowing if it was going to be possible to land the aeroplane manually, what downward view would be provided and so on. There was a school of thought that maintained that the pilot should not even try to take over from the automatic devices but that a sufficient number of channels should be installed so as to render pilot intervention unnecessary. He believed, however, that it should be possible to take over a landing at a late stage with the aid of the flight director.

It was suggested by a speaker from the Air Registration Board that a discussion on fuel systems would be useful. He thought it unlikely that any of the exotic fuels would be available in the early days of supersonic transport operations, although it would be nice to think of engines being fuelled with solid rods. Kerosine-based fuels would therefore form the bulk of the fuels available and he would like to hear views on the possibility of fuel leakage onto a hot aircraft and the possibility of spontaneous combustion.

Mr. Wilde reported that work was in hand on studies of kerosine-based fuels and those being used at present had a spontaneous ignition temperature of about 250°C—this would obviously present a problem in a $M=3{\cdot}0$ aircraft where typical temperatures were about 330°C.

Returning to the question of operating costs, a speaker from English Electric Aviation said that the two ways in which operators could keep the costs of supersonic aircraft to a reasonable level would be by reducing indirect costs and in achieving a high load factor. Had they done any work towards optimising the required capacity, taking into account the influence of departure time on the load factor?

Mr. Jackson said that indirect costs had reduced in recent years more rapidly than direct costs; they were pursuing studies to see what future trends were. The optimisation of required capacity was a difficult task and required for its advance further information from designers. In reply to another questioner he said that it was possibly regrettable for study of the supersonic transport that developments up to now had warranted the precedent of asking the passenger to pay less for improved service including travelling faster. He believed some people would pay more for speed, but if only 10 per cent of the people were prepared to do so this would not be enough to support the production and operation of a supersonic fleet.

Dr. Russell remarked that the question of economics was very complicated and would depend on a world-wide survey of operations. It was interesting to note that for two aircraft he had compared the actual air mile costs for the supersonic aircraft were lower than those for the subsonic one.

To bring the proceedings to a close, Sir George Gardner asked the co-Chairman, Mr. M. B. Morgan, to sum up.

Mr. Morgan thought that it had been a very profitable day. He was glad to find much of what he had said in a lecture to the Society in March 1960 endorsed by what had been said here.

On the aerodynamics side he would have liked to have heard more about stability and control but he gathered from the silence that this was a tractable area. The big lesson was that L/D was quite vital in this aeroplane. On the engine side it seemed that the engine people would have to be even more perfect than usual. From the materials angle it appeared that a Mach 2·0 aircraft was possible whereas a Mach 3·0 one looked awkward, especially on the non-metallic side, from the present stage of knowledge. It was now of vital importance to get the test techniques for light alloy structures sorted out in terms of testing hot and testing long, and it was obvious that much work would have to be done to satisfy the airworthiness authorities. On the propulsion side the arguments for turbo-jets and for turbo-fans were very closely run and the dominating factor seemed to be the noise they would be allowed to make on the ground. Contrary to the expectation some years ago, it seemed that these engines could be more acceptable in terms of noise level when climbing away from the runway than some existing jets. Noise on the ground was still a problem. The sonic boom story must be watched, but it was too early to be dogmatic.

From the piloting point of view, this could be termed a short range aircraft and prospective operators could learn from the present short haul operators. Just what went into the aircraft would depend on the present controversy about what the pilot would have to do, on how much of his time was to be spent in monitoring and how much in piloting. It did not seem that there would be anything especially new about the systems. Experience had already been gained with high subsonic jets which would help when it came to coping with the supersonic ones.

It had to be emphasised that this was still a very tender design and that the result of not being very clever would be even more disastrous than in the past. As against this the rewards of cleverness were higher. He stressed the importance of ground testing; more must be done on the ground, not just with structures but with the whole thing, to cut down the amount of proving time needed in the air.

Mr. Morgan also stressed the importance of making an overall attack on costs; the whole essence of the design was to get an economical supersonic aircraft. When estimating direct costs an enormous item was the initial cost of engines and airframe. The initial cost must not be so high that the aircraft, even if scientifically wonderful, was too expensive to operate.

In conclusion he said that the idea of a light alloy aeroplane cruising at about Mach 2·0 was attractive. Looking to the future it seemed that the end of the road had by no means been reached where subsonic aircraft were concerned and many things that they learnt from these would apply to the supersonic aircraft in due course.

A written contribution suggested that Dr. Russell could surely not take seriously, as representing current American thought, an unclassified NASA technical note which dealt with $M=3$ transports of canard design. The contributor wondered if Dr. Russell remembered the incident between the wars when the speed record on water was fought with keen rivalry, and on one occasion the American, Gar Wood, lured Segrave across the starting line after the second gun so that the race could be won by Gar Wood's reserve boat. Commercial rivalry could stoop to similar methods now, and in this country they must keep the dust from their eyes and remember that they could not lead the world on their hands and knees.

Dr. Russell replied that, while it was not impossible that the Americans were in a deep plot of deception, information from a wide range of sources was consistent with the unclassified NASA Technical Note. It would not be expected that reports of interest of opinion among the various constructors would be wholly consistent. There was, nevertheless, a strong indication for preference for $M=3$ transports of canard design. He personally found that conclusion difficult to understand.

A High Temperature Decompression Chamber for Developing Aircraft Environmental Control Systems

BY

O. D. FURLONG, A.F.R.Ae.S.

(Chief Development Engineer, Normalair Ltd., Yeovil)

1. Introduction

The greatly increased performance of both civil and military aircraft since the Second World War has raised problems undreamt of at an earlier period and, from the simple measures originally introduced to improve pilot's well-being, have sprung the complex installations which are now essential for human survival and comfort and to ensure the proper functioning of vital equipment carried in present day machines.

The term "Environmental Control System" is used for convenience to cover collectively the various cabin and equipment bay, pressure, temperature and humidity control systems that are fitted.

Early work in this field, which was conducted in the Experimental Engineering Department of Westland Aircraft Ltd., resulted in 1946 in the formation of Normalair Ltd. as a specialist company to expand these activities, and from the start it was apparent that to undertake adequate pre-flight development extensive test facilities would be required. A 3,000 cubic ft. decompression chamber complete with refrigeration system and internally mounted cabin specimen was brought into operation in 1945, and since that period the company's high altitude laboratory has been expanded progressively, so that at the present time, seven decompression chambers have been installed together with an oxygen laboratory, five cold air unit test cells and sections covering electronics, climatic, vibration and "*g*" testing, and instrumentation.

The largest and most complex single item of equipment is a decompression chamber, capable of operation at high temperatures, and which measures, internally, 42 ft. long by 15 ft. in diameter.

This paper sets out to discuss the design and layout of this chamber and associated installations and plant, with special reference to the problems of controlling the environmental conditions within the compartments of aircraft and testing the necessary systems.

As originally conceived the facility was designed with a view to meeting the requirements for research and development for a supersonic aircraft project but, following cancellation of contracts, this was curtailed at an early stage, and the work carried out since that time has not required the full temperature conditions or electric power resources catered for in the design. Since completion, however, the installation has had considerable use and, while security restrictions prevent all aspects of the work being published, it is possible to include some photographs and chamber operating data.

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2. Environmental Control System Testing

Aircraft installations can of course only be finally proved by flight trials, but to reduce the risk of failure in the air and so that the period of testing of modern aircraft can be kept to a minimum, it is of the utmost importance that the development of the numerous systems should be carried as far as possible before they are installed and operated in the machine.

With some types of equipment it is practical to employ an old aircraft as a "test bed" so that flight development can proceed before the prototype machine is ready. When considering the testing of environmental control systems however, which invariably have to be designed and/or set to meet the specific requirements for a particular type of aircraft, experience has shown that this method has limitations. Although useful work can be done in some instances, it is frequently found that installation problems and performance restrictions on the available machine prevent realistic operating conditions being established. The cost of such testing is also inevitably high and frequently proves out of all proportion to the usefulness of the results achieved.

Ground rigs are also only of limited value for, although they are relatively inexpensive to set up and it is a simple matter to install instrumentation and to make quick modifications which would be impractical in an aircraft, it is not generally possible to simulate the correct operating conditions covering the flight performance envelope.

The chamber under discussion has been installed to bridge the gap between conducting component development in the laboratory and carrying out aircraft flight testing. It was designed so that development can be carried out on complete environmental control systems under realistic flow, pressure, temperature and humidity conditions.

2.1. TYPICAL TEST CONSIDERATIONS

To establish the basic design performance of environmental control installations, it is the usual practice to select a number of representative flight cases and then to conduct a theoretical analysis of these.

There are obviously considerable advantages in selecting the same conditions as the basis of the Test Schedule so that a direct comparison can be made between the calculated results and the rig figures.

A number of "steady state" tests are therefore usually conducted on systems being developed in the chamber, *e.g.* at mean sea level, and at low altitude conditions runs would probably be carried out to simulate taxying, take-off, economic cruise, maximum aircraft speed and loiter speed conditions, while some of these and other appropriate cases would then be investigated at other fixed altitudes covering the whole performance envelope of the aircraft.

Because of the relatively long temperature and humidity time constants of most installations, care has to be taken during such runs to ensure that stable conditions have in fact been established throughout the whole system. Failure to do this can make analysis of the results difficult and unreliable. The flow and pressure time constants are on the other hand very much shorter and stable conditions are usually re-established quickly following a change.

At the conclusion of this series of tests, and after any necessary modifications have been carried out to ensure that the desired performance is obtainable, it is usual to conduct further simulated flights in which the behaviour of the system is studied during typical transient aircraft operating conditions, emergency cases and so on.

When setting up the chamber installation, certain assumptions must of course be made concerning factors dependent on the aerodynamic form of the aircraft, and both performance calculations and chamber test requirements have to be based on data supplied by the Aerodynamics Department of the Aircraft Manufacturer. Such considerations as the percentage ram temperature rise and the ram pressure recovery at the intakes, as well as the pressure variations over the aircraft skin, have to be taken into consideration. Wind tunnel results can often be used with advantage to simulate as accurately as possible realistic flight conditions.

One great advantage of setting up systems in the chamber in addition to proving them during aircraft development flying, is the relative ease with which the ducting and components can be instrumented. Piezometer rings and thermocouples are usually included in the ducting upstream and downstream of all components so that the pressures and temperatures can be obtained throughout the system. Pitot heads, yaw meters and so on, are also built in where necessary so as to allow local flow patterns to be explored.

Since the built-in measuring sections incorporated in the chamber installation allow a more accurate determination of the mass air flow being made than is usually possible in actual aircraft installations, the opportunity is often taken of calibrating sections of ducting and components destined for use on prototype aircraft by building them temporarily into the chamber system.

3. Aircraft Environmental Control System Requirements

Figure 1 gives a good general impression of the chamber and the main features of the layout are given in Fig. 2. Before discussing the specification in any detail, however, it is desirable to consider in broad terms the methods adopted for controlling the environment within the compartments of aircraft and the ambient conditions that have to be taken into account. It is not the purpose of this paper to analyse in detail the air conditioning load requirements or the mechanical details of the air conditioning equipment, but without some consideration of the matter it is impossible for those not familiar with the subject to appreciate fully many aspects of the design of the facility and associated plant installations.

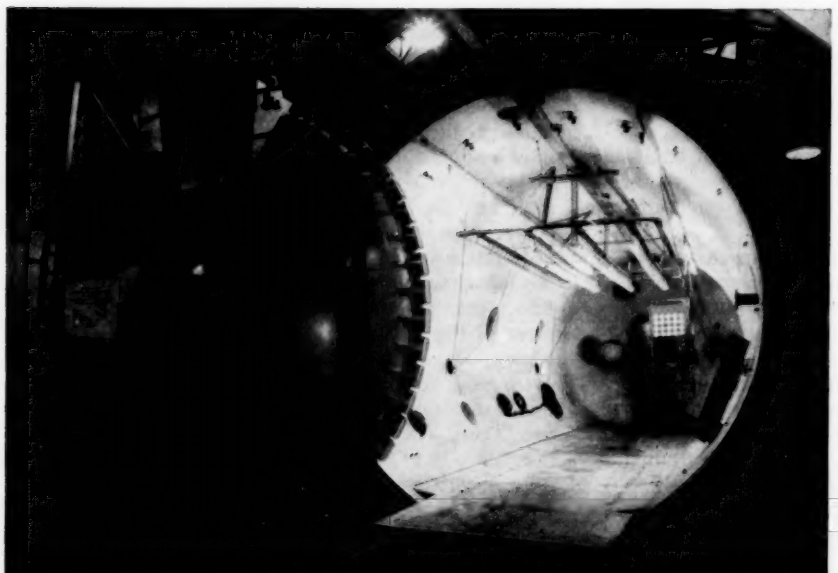
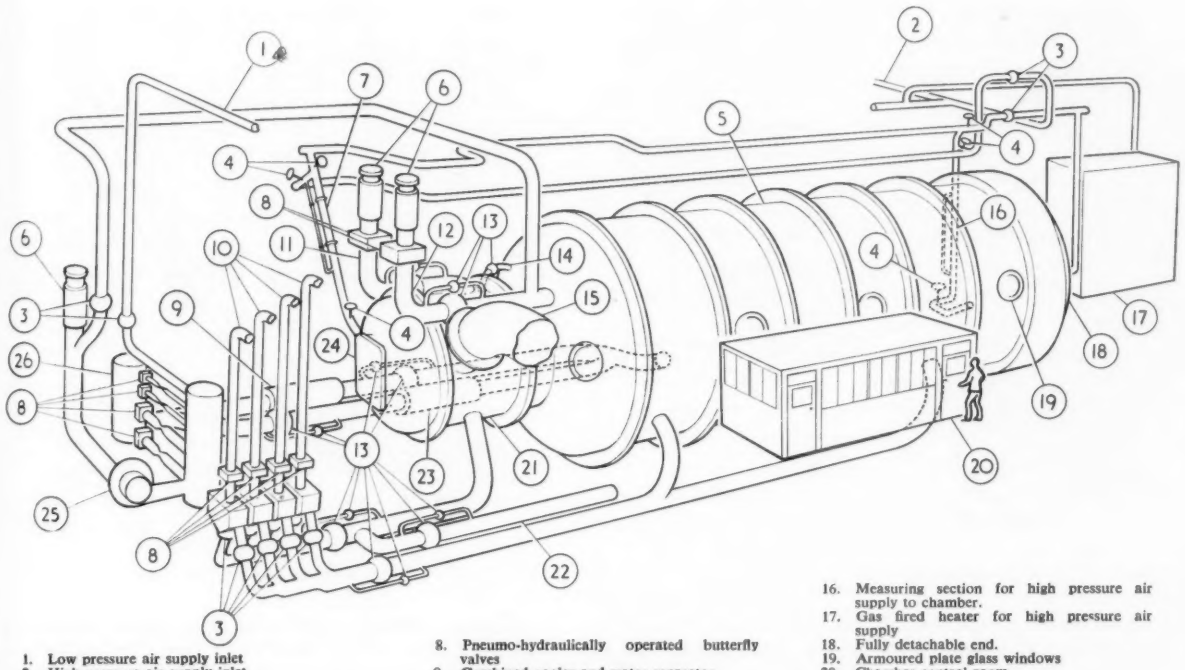


FIGURE 1. General view of the chamber.



1. Low pressure air supply inlet
2. High pressure air supply inlet
3. Hand operated valves
4. High pressure supply automatic control valves
5. Chamber
6. Silencers
7. Measuring section, high pressure supply to airlock
8. Pneumo-hydraulically operated butterfly valves
9. Combined cooler and water separator
10. Outlet to exhausters
11. Bleed to airlock from ambient
12. Bleed to chamber from ambient
13. Electrically operated gate valves
14. Bleed pipe between airlock and chamber
15. Interconnecting pipe to other decompression chambers
16. Measuring section for high pressure air supply to chamber.
17. Gas fired heater for high pressure air supply
18. Fully detachable end.
19. Armoured plate glass windows
20. Chamber control room
21. Airlock
22. Cooling air bleed to exhauster line
23. Electric air heater
24. Airlock door
25. Centrifugal fan
26. Measuring section for low pressure air supply

FIGURE 2. Diagram of chamber installation.

3.1. DESIGN PARAMETERS

The parameters which have to be borne in mind when assessing the performance and test requirements for such aircraft systems can be conveniently divided into two main groups, namely: —

(*a*) External conditions, such as ambient air pressure, temperature and humidity, and the effects of air speed and solar radiation.

(*b*) Internal considerations, such as the volume of the cabin, the number of crew and passengers carried, the quantity, pressure and temperature of air available for cabin conditioning, the size and location of transparencies, the thermal insulation of the cabin and the heat output of electrical equipment carried.

This list, which is by no means exhaustive, serves to show the complex nature of the problem, particularly when it is remembered that many of the above so-called internal factors are themselves either directly, or indirectly, influenced by the external conditions. Account must also be taken of the response rates and thermal capacities of the various components in the system and the transient effects that these have in the rapidly changing conditions experienced during flight.

3.2. ENVELOPE OF EXTERNAL CONDITIONS

Most modern high performance aircraft are designed for world-wide coverage, thus the environmental control systems must be designed to function adequately over the whole range of ambient pressure, temperature and humidity conditions met at any part of the globe during flights at any altitude and speed at which the aircraft is capable of operating. These factors will therefore be briefly considered first.

The reduction in atmospheric pressure which occurs with altitude needs no discussion here but, Fig. 3 shows, in addition to the ICAN temperature changes, the envelope of static ambient temperature conditions/altitude normally used as a basis for civil design work in this country[1]. However, since experience has shown that more severe conditions are in fact found in some tropical areas, a temperature of 50°C is often used for calculating maximum cooling loads in the ground layer.

Figure 4 shows the ram air temperature rise/true air speed in knots, and indicates the magnitude of the kinetic heating problems and how the direct cooling potential is reduced as aircraft speeds are increased.

At the present time there is no single generally agreed specification covering world-wide humidity/temperature conditions to be used for aircraft design purposes, but Fig. 5 shows an envelope of conditions based on tables published by the Meteorological Office[2].

In the past, failure to appreciate fully the important part that humidity plays in environmental control work has led to disappointing results, and it will be seen later that special measures have to be taken when conducting system tests to ensure accurate measurement and control of humidity.

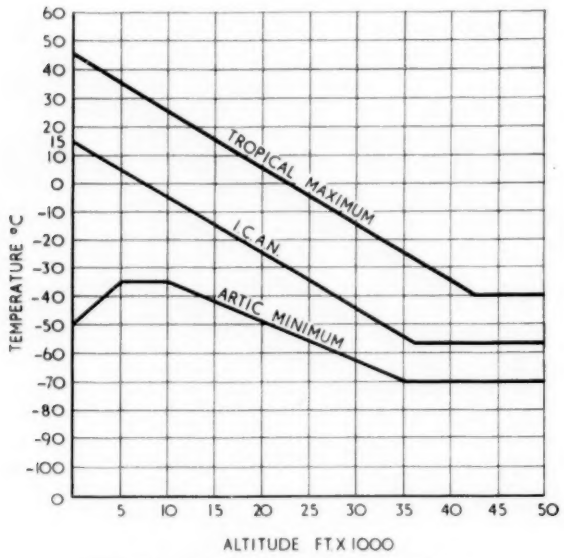


FIGURE 3. Temperature/altitude conditions.

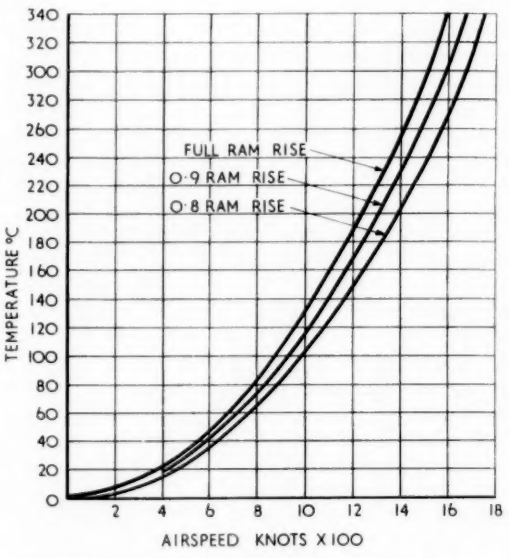


FIGURE 4. Ram temperature rise/air speed in knots.

The effects of solar radiation can appreciably increase the cooling load for an aircraft cabin, particularly when stationary on the ground and at altitude in tropical latitudes, this factor must therefore not be neglected. The solar constant is usually taken as being 4 C.H.U./min./ft.^{2} but this value is reduced considerably by such factors as the humidity and dust content of the air, the solar angle, the projected area of the surfaces normal to the sun as well as the transmission factor of the transparent material.

As an example of how in flight altitude actually affects the values, two typical results are quoted from the heat balance of a machine operating at mid-day under clear sub-tropical conditions. These show that at low altitude solar radiation accounted for a cooling load of 2,450 C.H.U./hour, while at 40,000 ft. this value had increased to 3,100 C.H.U./hour.

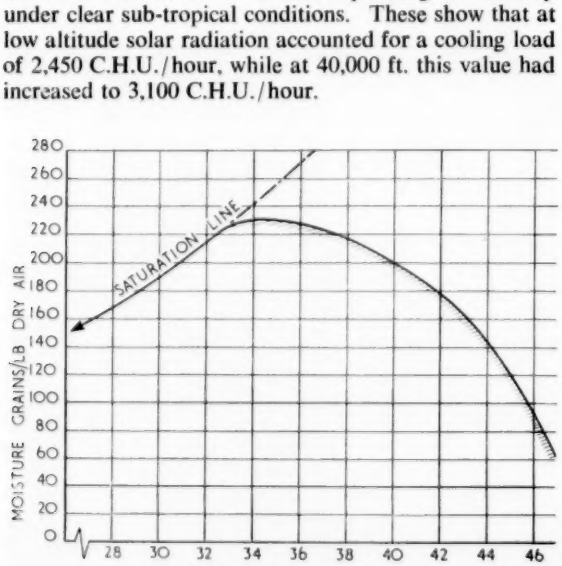


FIGURE 5. Envelope of humidity conditions.

Having briefly catalogued the most important external factors which have to be borne in mind, it is now possible to look more closely into the requirements for typical aircraft systems.

3.3. TEMPERATURE CONTROL

Whether the air supplies needed for aircraft compartment pressurisation and air conditioning purposes are tapped from the main engine compressors, or are obtained from mechanically driven cabin superchargers, the heat of compression is such that in very few present day installations is any additional source of cabin heating required once the machine is airborne. The chief exception to this is in the case of some civil aircraft where local warmth is provided by fitting electrically heated carpets and panels. In most cases, however, these are used to counteract local variations in cabin temperature distribution rather than to contribute to any major exent to the overall heating load.

Under many conditions of flight, some degree of cooling is necessary and while ram air heat exchangers often suffice at lower speeds and high altitude conditions, the effect of cabin air supply compression ratio, ram air temperature rise, internal heating loads and so on, makes the use of additional refrigeration essential even under conditions when the ambient air is well below freezing.

Both air cycle and vapour cycle cooling systems, used either by themselves or supplementing each other, are currently used, and Figs. 6 and 7 illustrate diagrammatically typical systems of each type. In some installations on high speed aircraft, the fuel is used as a heat sink instead of ram air, and air/fuel heat exchangers are fitted. Under supersonic conditions, where temperatures are very high, these systems often have to be further supplemented by evaporative cooling methods, water boilers being installed in some machines of advanced design.

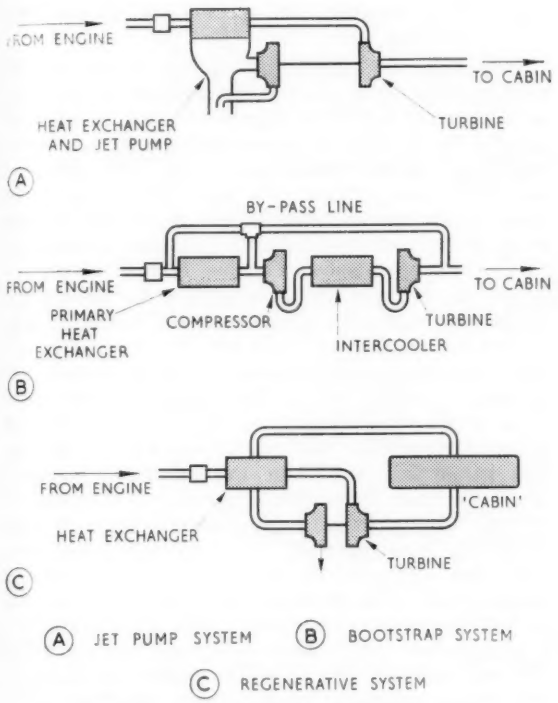


FIGURE 6. Typical air cycle cooling systems.

The actual air-conditioned compartment temperatures in present day aircraft are usually automatically as well as manually controlled and electric, electronic and pneumatically operated regulating systems are all in current use.

3.4. HUMIDITY CONTROL

As previously mentioned the humidity content of the air used in aircraft conditioning systems is a matter of considerable importance, not only because of the effect it has on the cooling capacity required, but also because of the added complication it causes to the design of environmental control systems.

Very low relative humidity in an aircraft cabin can cause discomfort to the air crew and passengers, so that humidifiers are fitted in many civil aircraft which cruise for long periods at high altitude.

At the other extreme, the high relative humidity often found at low altitudes (*see* Fig. 5) can be a serious embarrassment as it can cause water precipitation, fog and even snow within the cabin, as well as icing in the system and misting of the transparencies. To help overcome this, water separators have to be fitted in practically all modern civil and military aircraft, and the lower temperature limit of the air used for cabin cooling is often dictated by the problem of icing.

3.5. PRESSURE CONTROL

So far only problems associated with temperature and humidity control have been discussed, and it is now necessary to consider briefly some aspects of the

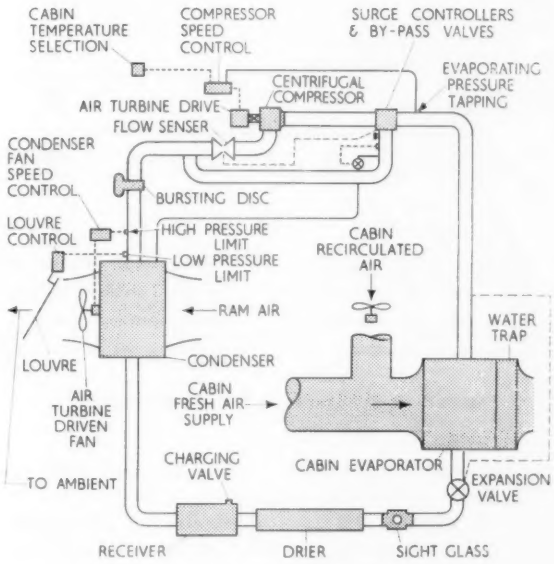


FIGURE 7. Typical vapour cycle cooling system.

performance and testing of aircraft pressure control equipment.

Weight penalties, and in military aircraft the risk of explosive decompression under combat conditions, make it undesirable for sea level pressure to be maintained in the cabin throughout flight, although physiologically this would of course be the ideal solution.

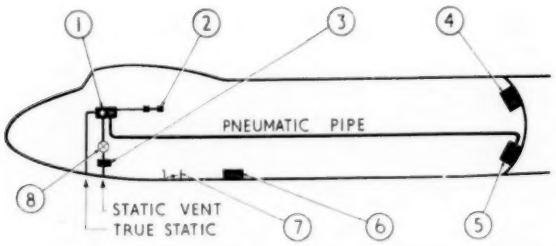
The equipment therefore has to be designed to regulate the pressure in the most advantageous manner within the limits dictated by the aircraft performance and the maximum designed cabin differential pressure.*

In the case of civil aircraft the normally accepted maximum cabin altitude is 8,000 ft. and, because of passenger ear and sinus troubles, rates of change of cabin altitude have to be kept as low as possible at all times, 300 ft./min. being regarded as desirable, although the performance of some aircraft make somewhat higher values necessary.

In military machines cabin altitudes of up to approximately 25,000 ft. are permitted when oxygen breathing equipment is being used and this altitude can be increased to around 40,000 ft. by resorting to pre-oxygenation of the crew. At altitudes above this some form of pressure garment has to be worn. The rates of cabin altitude change cannot of course be kept at anything approaching the low values required on civil aircraft, but the matter is still of the utmost importance, and has to be considered carefully in each design.

Automatically controlled discharge valves (*see* Fig. 8 for typical pressure control system) are used universally on all present day pressurised aircraft to regulate the outflow of air from the cabin, thus retaining the desired cabin pressure independently of the inlet flow used for conditioning purposes.

*Cabin differential pressure = cabin absolute pressure − aircraft ambient absolute pressure.



1. Automatic pneumatic pressure controller with adjustable cabin altitude and rate of change selectors
2. Inlet filter and moisture condenser
3. Emergency pressure controller
4. Combined safety and inward relief valve
5. Combined discharge and inward relief valve
6. Ground pressurising connection
7. Dump valve
8. Manual over-ride valve to shut discharge valve

FIGURE 8. Typical pressure control system (civil).

Once the aircraft performance, the envelope of cabin flows and so on, are known, it is possible to carry out most of the development of this equipment independently from the rest of the cabin environmental control installation. However, in the final stages, it is often advantageous to be able to test the combined systems together, so as to check the overall response and stability.

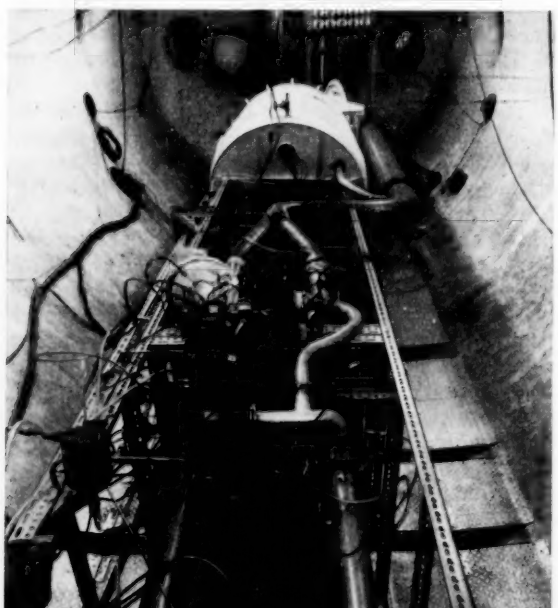


FIGURE 9. High speed fighter environmental control system in chamber.

3.6. TYPICAL CHAMBER INSTALLATION OF AN AIRCRAFT SYSTEM

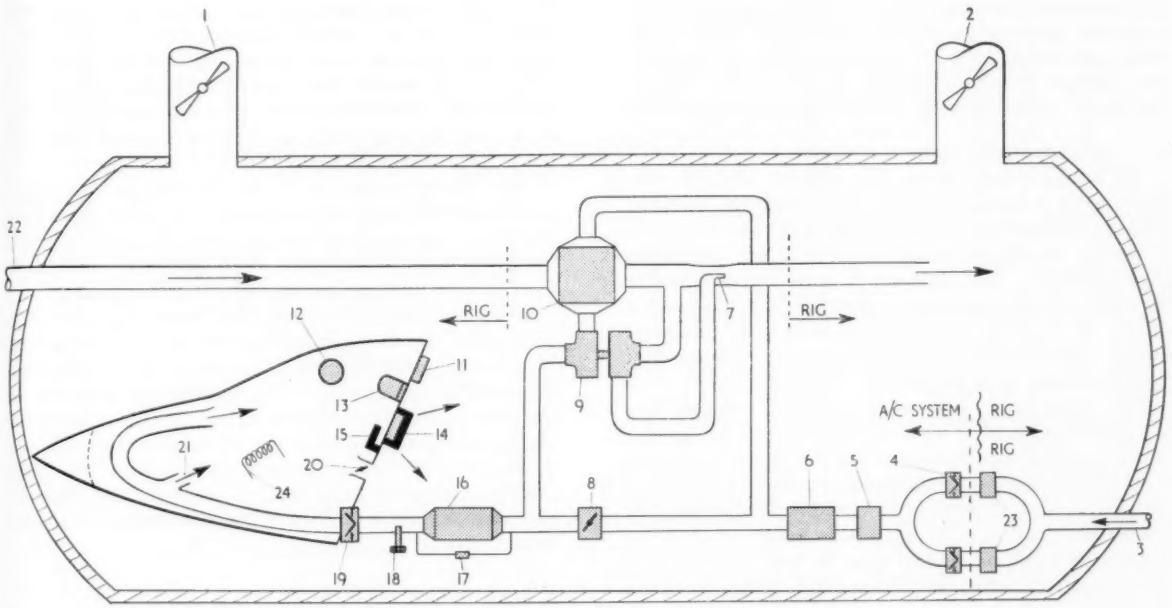
The range of aircraft installations which can be usefully subjected to chamber development is obviously so wide that it is difficult to generalise but, as a typical example of the type of system tested, Fig. 9 shows a high-speed fighter installation. Fig. 10 shows a simplified layout of this. An important feature to be noted is that two independent pre-conditioned sources of air have to be supplied to the chamber to simulate the engine tapped and ram cooling flows required.

As can be seen on the diagram, the air for the system first enters the installation via a bifurcated pipe to simulate tappings from the two aircraft engines. It then passes through non-return valves which are installed to prevent reverse flow occurring in the event of an engine being shut down. Next it passes through an electro-pneumatic shut-off valve which can be controlled from the cabin. From this unit the air is ducted to an automatic flow controller which is designed to prevent over-speeding of the cooling turbine at high engine pressure conditions. Downstream of the automatic flow controller, the ducting is again divided and the air passes either through a heat exchanger and cold air unit to supply cooling to the cabin, or direct through a temperature control valve providing heating. The two lines rejoin downstream of the turbine and upstream of a water separator which extracts any free water present. A high-limit thermostat is situated downstream of the water separator, and a differential pressure controller tapped across the water separator is fitted so as to prevent excessive pressures building up in the event of ice blocking this unit. After passing through a further non-return valve mounted on the cabin wall the air is finally passed into the cabin via the distribution ducting.

The ram air is ducted from the intake to the air/air heat exchanger which cools the engine bleed supply to the cold air unit. From there a tapping is taken to load the compressor of the cold air unit. The outlet from this is ducted to the nozzle of a jet pump, the purpose being to induce cooling air through the heat exchanger at low aircraft forward speeds when the ram pressure is low.

The proportions of the hot and cold air flows to the cabin are regulated by the temperature control valve, which responds to pressure signals received from the temperature sensing elements and the differential pressure controller. The main sensing element situated so as to be in the air flow through the cabin discharge valve has incorporated a bi-metallic element which controls the operating pressure to the temperature control valve by means of a small pneumatic servo, the datum position of this being set by a temperature selector in the cabin. The two other elements, *i.e.* the high-limit thermostat and the differential pressure controller, are not normally in operation but can override the main element if required.

The selection of cabin temperature can be either manual or automatic, automatic control being by means of the sensing element described above, while manual selection gives the pilot direct control of the temperature



PNEUMATIC SERVO CONTROL AND
STATIC VENT LINES ETC., NOT SHOWN

1. To exhausters
2. Chamber ventilating bleed
3. Compressed air inlet
4. Non-return valve
5. Electro-pneumatic shut-off valve
6. Automatic flow controller
7. Jet pump
8. Temperature control valve
9. Cold air unit
10. Heat exchanger
11. Combined safety and inwards relief valve
12. Differential pressure selector
13. Pressure controller
14. Discharge valve
15. Sensing element
16. Water separator
17. Differential pressure controller
18. High limit thermostat
19. Non-return valve
20. Depressurising valve
21. Distribution ducting
22. Ram inlet
23. Rig shut-off valves
24. Electric element to simulate cabin heat loads

FIGURE 10. Diagram of simplified fighter installation.

control valve which will then only respond to his operation of the control knob.

The diagram also shows the discharge valve mounted on the cabin bulkhead, which in conjunction with the pressure controller automatically regulates the cabin pressure as the aircraft climbs or descends. A combined safety valve and inwards relief valve for emergency purposes can be seen as well on the bulkhead. A cabin differential pressure selector is provided so that the pilot can over-ride the automatic pressure controller and obtain an alternative low cabin differential pressure selection. Finally, a depressurising valve is fitted, which can be controlled by the pilot to vent the cabin to ambient if required.

Experience has shown that when conducting chamber development on systems of comparable complexity to the one discussed above, considerable practical advantages are obtained by rigging the complete installation and cabin specimen on a wheeled framework. Such an arrangement allows initial construction and subsequent modifications to be undertaken without keeping the chamber out of action for long periods. The rationalisation of all instrumentation is also of great importance, and wherever possible instrument leads, pipes and so on, are taken out of the chamber via a single easily removable adaptor plate.

4. Design and Construction of the Chamber and Associated Installations

Having discussed in general terms the requirements for aircraft environmental control and system testing, it is now possible to consider in more detail the design of the decompression chamber.

The unusual requirements laid down in the specification raised many difficult structural and mechanical design problems, and it would be possible to devote a paper entirely to discussing the solutions found. However, since these have indirect bearing on the use of the facility for system testing, only brief reference can be made to them. Consideration is given primarily to those aspects of the design which decide its usefulness as a tool for research and development purposes.

Following the preliminary investigations and design studies which were carried out at Yeovil so as to clarify the basic requirements, the work of construction was undertaken by Markham and Co. Ltd. of Chesterfield. This firm undertook the design, fabrication and erection of the vessel on site, up to the stage where it was successfully pressure and vacuum tested.

Once the chamber was in position and tested, the piping, lagging, control systems, electrical power supplies, instrumentation and so on, were designed and installed by Normalair personnel and by sub-contractors

working directly for the company. The single exception to this was the large gas fired air heater (*see* Section 4.5), which was designed and built by Industrial Furnaces Ltd., Kingswinford, on behalf of the South Western Gas Board who took responsibility for this installation.

4.1. SITE CONSIDERATIONS

Since the whole project was designed and constructed as a private venture, the need for financial economy dictated that, as far as possible, the maximum utilisation should be made of the existing laboratory plant facilities. This decision in turn severely limited the choice of site, and made it essential that the chamber should be located in juxtaposition to the main exhauster installations, thus reducing the vacuum line pressure losses to a minimum.

The most satisfactory solution to this problem was to enclose a section of 40 ft. wide roadway, running between the main laboratory plant house and the building in which the firm's cold air unit test cells and associated compressor installations are housed. By locating the installation in this position, it was possible to ensure not only the minimum pipe runs to both plant houses, but also direct access to the chamber. This has since proved of great convenience when off-loading heavy equipment and preparing for tests.

4.2. GENERAL CONSTRUCTION

Examination of the site revealed that the foundations would be in compacted blue clay, and to achieve the necessary stability, it was decided to support the vessel on a 59 ft. long reinforced concrete raft. To ensure maximum strength, this was laid down in a single pouring and some 3,000 cubic ft. of concrete, weighing approximately 200 tons, was used.

The chamber and airlock, which are designed for an internal pressure range from full vacuum to 15 lb./in.^{2} gauge, were prefabricated in five sections at Chesterfield and transported by road to Yeovil, where they were welded together on site. Three-quarter inch steel plate is used throughout, except for the domed ends, which are $1\frac{1}{8}$ in. thick. The barrel of the main vessel is reinforced by seven external T-section hoops, which are spaced at approximately equal distances along its length.

The airlock which is 10 ft. long and 7 ft. 6 in. diameter (inside lagging) is welded concentrically to one of the domed ends of the chamber. Its barrel is strengthened in a similar manner to the main vessel by two hoops.

Altogether eight 20 in. diameter armoured plate glass windows are provided in the chamber walls, three being situated on each side, and two in the domed end next to the airlock. One similar window is provided in the side of the airlock. All the windows are attached by special V-flanges and clamps, which allow quick removal, and for very hot runs lagged steel plates can be fitted in place of the glass.

The total weight of the metal work in the combined chamber and airlock is approximately 92 tons, and to cut down the effects of heat capacity to a minimum, both compartments are internally lagged to a thickness of 6 in. with Limpet Asbestos, this is faced for protection with a layer of Bulldog composition. Calculations show that when the windows are replaced by the lagged adaptors, the overall heat loss through the walls is approximately 450 C.H.U./min., when the vessel is soaking at an internal temperature of 250°C (maximum designed temperature) and the ambient is at 15°C.

As shown in Section 5, the problem during many test conditions is one of heat dissipation rather than heat retention, but structural requirements and the time lags involved, make it desirable that the steel structure is kept as near ambient temperature as possible.

Two pressure-tight doors, each measuring 5 ft. × 3 ft. with 6 in. corner radii, are provided to allow access into the airlock and from the airlock to the chamber. A small manhole 2 ft. 6 in. diameter is provided in the domed end opposite the airlock to allow entry for rigging purposes after systems have been installed. In addition to the windows mentioned above, two large adaptor plates are fitted into the chamber and one into the airlock; these are used for instrumentation and other services.

4.3. HANDLING FACILITIES

Careful consideration was given to the matter of designing the installation so as to facilitate the rigging and dismantling of systems with the minimum of labour and time delay. It was decided that the domed end opposite to the airlock should be completely and easily removable, and that the internal working level should be the same as the floor level of the building.

The floor is made of light alloy plates which are capable of taking a mean load of 100 lb./ft.^{2} It is constructed in the form of easily removable panels and, immediately below, two 56 lb. bridge rails are fitted at 5 ft. centres. These run the full length of the vessel and extension lengths are carried out to some distance beyond the building. Heavy equipment of up to 12 tons dead load can thus be mounted direct onto trolleys and run into position with the minimum of effort.

A further aid to handling is provided in the form of an overhead 6 in. × 5 in. R.S.J. rail mounted on the vertical centre line of the vessel. This also extends on gantries to a point well beyond the building and, if desired, distributed loads of up to $2\frac{1}{2}$ tons can be slung in the chamber by this means. All the above features are illustrated very clearly in Fig. 1.

4.4. EXHAUSTER CAPACITY

Both the chamber and airlock are piped into the main laboratory exhauster system and the layout is shown diagrammatically in Fig. 2.

The exhauster installation is comprised of eight electrically-driven Hick Hargreaves R.V.17 pumps, which are arranged in four sets. To improve the performance under high altitude conditions, these can be backed by smaller second stage machines. The single and two-stage flow/altitude envelope curves for the plant are given in Fig. 11.

To prevent over-heating of the pumps during hot runs, water-cooled Serck Heat Exchangers are built into the pipe lines running to the four exhauster sets. The

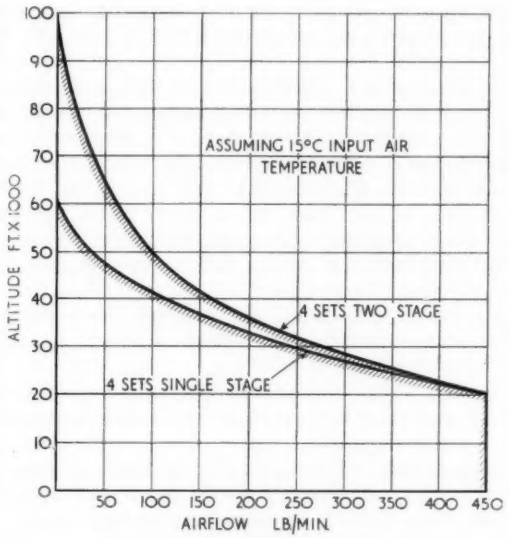


FIGURE 11. Capacity of exhausters.

coolant flow through each is 2,000 gal./hr. and the overall design heat extraction rate of the installations is 420,000 C.H.U./hr. when cooling exhausted air from 250°C down to 35°C. Care was taken in the design of these units to ensure that the pressure loss through them is kept to an absolute minimum, as any restriction to the air flow can, of course, result in a considerable reduction in exhauster capacity at high altitudes.

4.5 HIGH PRESSURE AIR SUPPLIES

One of the most important factors when considering the supply of air for system test purposes is its cleanliness and water content. Oil carry-over not only contaminates rigs and invalidates results, but it also constitutes a very serious fire and explosion hazard, particularly when conducting high temperature tests.

Lack of knowledge concerning the water content of the air during any particular test can also make analysis of the results difficult and in some cases impractical to undertake. It is therefore most important that careful control over humidity should be exercised at all times.

It is usual practice initially to calculate the performance of cabin conditioning systems on the basis of "dry air rating" and then to apply corrections to establish the effects of operating under other expected humidity conditions. Likewise, it is often desirable to conduct tests under "dry air rating" conditions and then to carry out repeat runs at different humidity values. This implies that the plant installation supplying air to the chamber must include means of accurately controlling its humidity content. In view of the large quantities involved, and the duration of some tests, the only really successful method of achieving this is to lower the air temperature to just above freezing point, so that practically all the water vapour precipitates out. The air then has to be reheated to the required temperature and means have to be provided for injecting

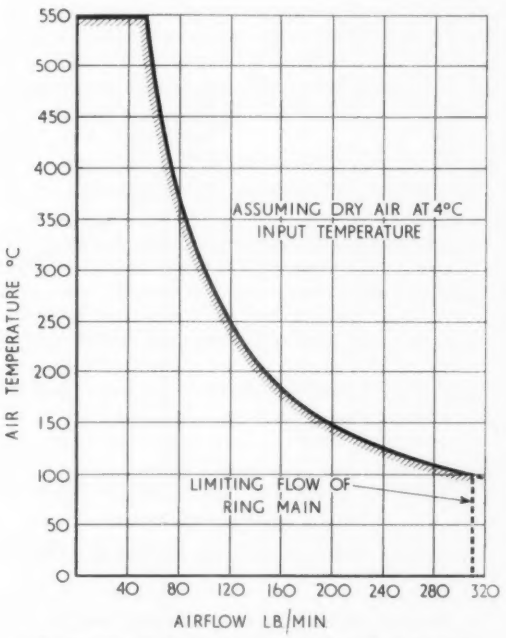


FIGURE 12. Performance of compressed air heater.

water at accurately known rates into the system. To ensure complete vapourisation, this is best done at the hottest point in the rig, *i.e.* just downstream of the heater.

The nine compressors which feed the laboratory ring main have a total output of 310 lb./min. at a maximum pressure of 150 lb./in.^{2} gauge, and special precautions are taken to ensure its cleanliness and controlled humidity. On leaving the compressors, the air is passed through after-coolers, filters and capacity tanks and, before entering the main is cooled further to a thermostatically controlled temperature of approximately 3°C. The precipitated moisture is then drained off so as to ensure that the maximum water content does not at any time exceed 11 grains/lb.

The total output of this ring main could, if necessary, be fed to the chamber, although the occasion has not yet arisen when this has actually been required.

As an alternative arrangement, the output of two of the pumps (58 lb./min.) can be tapped directly to the chamber and airlock at a maximum supply pressure of 200 lb./in.^{2} gauge.

The actual pressure of the air supplied to systems under test can be regulated to any required value by Honeywell Pressure Controllers.

The performance of the large gas-fired heater previously mentioned is given in Fig. 12 and as this is situated very near the vessel, the curve effectually shows the flow/temperature envelope of the high pressure air which is available for system testing.

The heater itself can be automatically temperature controlled, but because of the need for rapid response, and the large thermal capacities involved, cold air mixing systems are installed downstream of the heater which

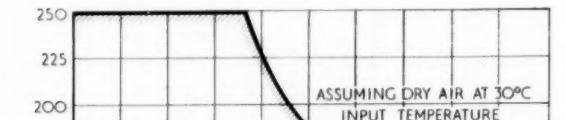


FIGURE 13. Performance of low pressure air heater.

can be manually or thermostatically regulated by means of Honeywell Temperature Controllers.

British Standard Nozzles are used for flow measurement, and a water injector is fitted in the line for humidity control. To prevent corrosion all the hot high pressure pipes are made of stainless steel, and they are heavily lagged to prevent heat loss.

4.6. LOW PRESSURE AIR SUPPLIES

When operating at altitude, air required for ram and ventilating purposes can be induced into the chamber, but at ground level and when simulating high speed flight conditions at low altitudes it is necessary to supply air under positive pressure.

To meet these requirements the installation is piped into the laboratory low pressure air system, which is fed from eight Reavell Compressors with a total output of 240 lb./min. at a pressure of 12 lb./in.^{2} gauge. In addition a general purpose centrifugal fan capable of supplying 130 lb./min. of air at a head of 55 in. W.G. is also tapped into the ram air supply duct so, if necessary, a total flow of 370 lb./min. can be supplied at approximately 2 lb./in.^{2} gauge.

So that the total ram flow can be established a special measuring section incorporating a range of British Standard Nozzles is built into the supply duct, and in order to control the air to the desired conditions, a large heater, cooler, water separator and water injector are also installed. The performance envelope of the heater which has a maximum output of 234 kW. is given in Fig. 13.

The cooler is made in two sections with the water separator mounted between them. The first stage is used to lower the air temperature to just above freezing point, so that free moisture can be collected in the separator, thus preventing the possible icing of the second stage during long runs. Brine at $-35°C$ is used as a coolant, the maximum flow being 600 gal./hr. The total capacity of the system is such that 240 lb. of air/min. can be cooled through 36°C.

4.7. ELECTRICITY SUPPLIES AND INFRA-RED HEATING

To meet various requirements for different aircraft systems and test conditions, the installation is wired for the following electric power supplies:—

(*a*) 18-32 V. d.c. up to 4·8 kW.
(*b*) 200 V. a.c. 3 ph. 400 c./sec. 1 kVA. (40 kVA. can be provided if required for special purposes).
(*c*) 0—262 V. 9 independently tapped phases, total output 351 kVA. at 50 c./sec.
(*d*) 230 V. a.c. Single ph. 50 c./sec Twelve 5 amp. socket outlets are available in the chamber for general use.
(*e*) 200 V. d.c. up to 100 kW. is available in the vicinity of the vessel and this could be connected up if required.

Removable strip lighting is installed in both the chamber and airlock; this can be taken out before the start of hot runs. For safety, the lighting circuit and all other 230 V. 50 c./sec. supplies are contactor controlled and can be cut off instantly at the console if required.

In the case of (*c*) three 3 ph. star connected transformers have been installed to provide a 552 tap variable voltage supply for heating and other purposes. Normally two of the transformers are used in conjunction with the 234 kW. ram air heater, while the third is connected into the vessal for general use.

Although the design soaking temperature of the chamber is limited to 250°C, far higher temperatures can be obtained on equipment mounted inside by using 1,000 W. Quartz tube heaters in conjunction with suitable reflectors. With this arrangement intensities in the order of 7·5 Watts/cm.^{2} can be obtained at a colour temperature of approximately 2,500°K. If necessary, the whole output of all three transformers can be used for this purpose, the installation being under the finger tip control of an operator seated at the console.

During many less severe tests, where it is required to operate systems under hot conditions, arrays of 250 W. and/or 375 W. infra-red lamps are used. Such an arrangement can be clearly seen in Fig. 1.

4.8. CONTROL CONSOLE AND INSTRUMENTATION

To facilitate ease of operation, the chamber controls and instrumentation are located on a two-position console (*see* Fig. 14). The operator on the right has control over the pressure and air flows, while the man on the left is mainly responsible for regulating the temperature conditions.

Duplicated panels carrying chamber, airlock and cabin specimen altimeters, rate of climb indicators and thermometers, keep both the operators informed of the general conditions, and an aircraft type intercommunication system allows them to communicate as required with other test personnel both inside and outside the vessel.

The push button controls for regulating the output voltages of the three transformers discussed in Section 4.7 can be clearly seen on the left hand of the photo-

graph and the layout of the air control valves operated from the console, is as shown in Fig. 15. It will be noted that the essential valves are all duplicated to ensure safety when working with personnel in the chamber at high altitudes. Two completely different types of controls are used, one set consisting of electrically operated gate valves, while the other uses pneumo-hydraulically operated butterfly valves. Emergency action can therefore be taken even in the event of complete failure of either one of these systems.

In the original design it was considered that the butterfly valves would be used purely for emergency purposes, but experience has shown that for some requirements they have better characteristics than the electric powered controls, and they are favoured by the operators for certain test work.

As shown on the photograph, the gate valves are operated by push-button controls, similar to those used for regulating the transformers, and the butterfly valves by engine throttle type levers. The traversing speed of the main 14 in. gate valves is 22 in./min. and that of the 3 in. by-pass gate valves 11 in./min.

In the case of the butterfly valves it has been found possible to regulate the full travel times between 10 seconds and 1 minute, by varying the pneumatic supply pressure.

The recording Honeywell pressure and temperature controllers used for regulating the conditions of the high pressure air supply, can also be seen clearly in the photograph; they are mounted on the back panel of the console.

For air flow measurement a bank of four 30 in., high pressure manometers (one mercury and three water) are used, together with the nozzles fitted into the high pressure and ram air supply lines. Pressure gauges are also installed for determining the absolute pressure at the measuring points, the air temperature being measured by Sifam temperature indicators.

An indication of the humidity content of the high pressure air entering the chamber is obtained by a Foxboro-Yoxall "Dewcell" indicator. In general, however, during "steady state" tests, the humidity content of the air piped into the installation is obtained by direct measurement of the injected water supply, as this is found to be the most reliable method.

In addition to the basic instrumentation already mentioned, a certain amount of equipment and measuring facilities are permanently built into the console for the instrumentation of systems undergoing test, and the most important items are as follows:—

Static Pipe Connections

(*a*) 34 permanent $\frac{1}{8}$ in. diameter pipes are led from the console to the chamber for taking pressure readings.

(*b*) 17 similar pipes are taken to the airlock.

Electric Ways

40 general purpose 5 amp. capacity electric ways are run from the console to the chamber and a similar number are connected into the airlock.

Thermocouple Ways

(*a*) 30 thermocouple ways are taken into the chamber.

(*b*) 20 thermocouple ways are taken into the airlock.

Temperature Recorders

(*a*) One six-channel Honeywell strip chart temperature recorder range 0–1,200°F (648°C). This can be connected up as required.

(*b*) One 12-channel recorder similar to that above, range 0–200°F (93°C) is also installed.

Temperature Indicators

Two Sifam temperature indicators range 0–600°F (315°C) are installed. These instruments are used mainly to measure the bearing temperatures of cold air units or Freon compressors incorporated in systems.

Speed Measurement

One Phillips Oscilloscope and R.C. Generator (Oscillator) range 20 c./sec.–20 kc./sec. This equipment is usually used in conjunction with electro-magnetic pick-up coils for measuring the speed of cold air units and Freon compressors.

FIGURE 14. Control console.

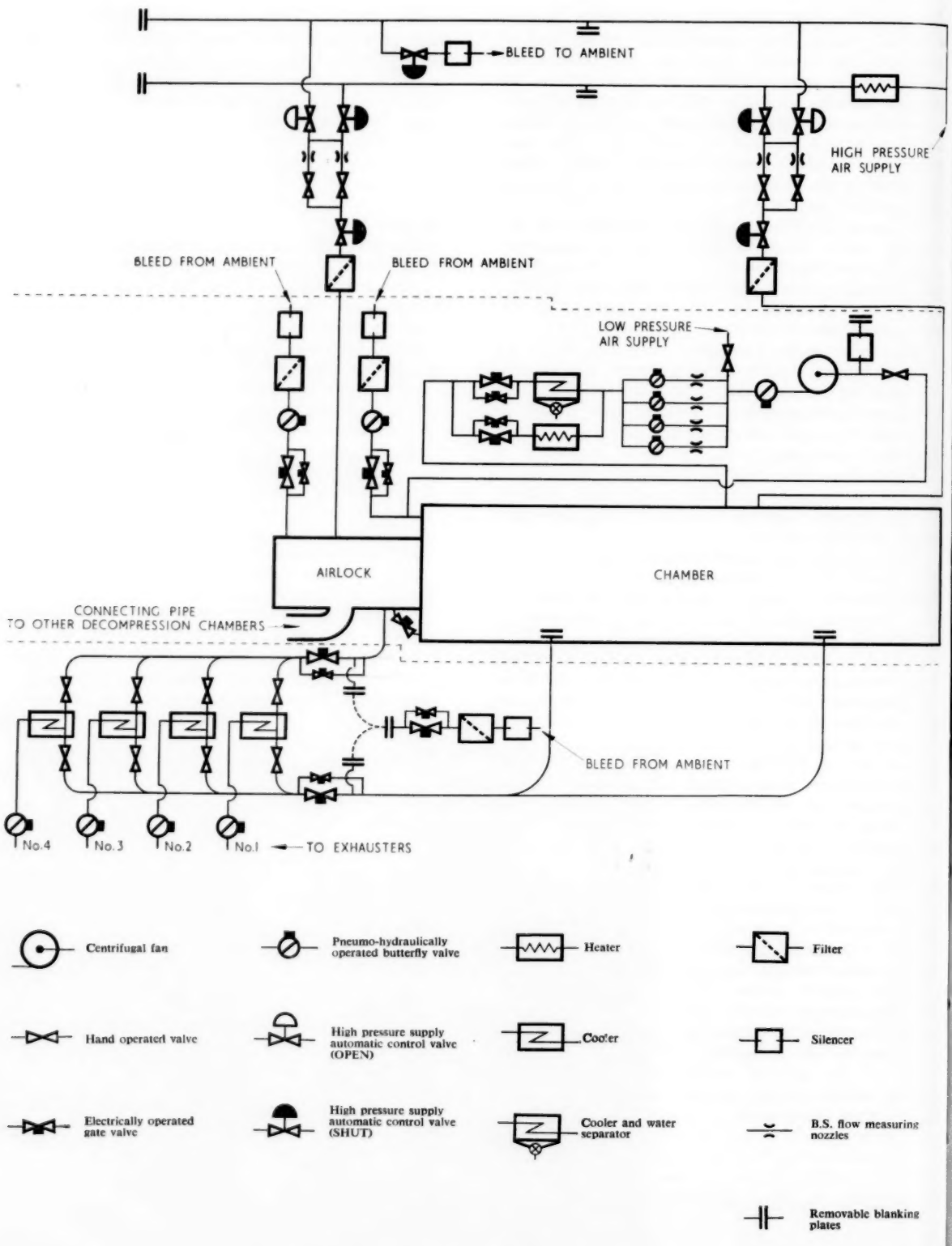


FIGURE 15. Diagram of chamber controls.

Cooling Water Flow Measurement

One 1,200 to 6,000 gal./hr. Rotameter is provided in the water main for measuring the total cooling water flow to any rig heat exchangers that may be used in systems.

In addition to this permanent equipment mounted on the control console, temporary panels are of course rigged when required to carry additional instrumentation necessary for particular test installations. Any special adaptor plates to carry the necessary leads out of the chamber can often be fitted in place of one of the armour-plate glass windows.

When conducting simulated flight tests which involve the exploration of transient conditions a 12-channel N.E.P. direct writing galvanometer recorder is frequently used in conjunction with suitable transducers.

5. Consideration of Some Test Installation Problems

It is self-evident that if an aircraft system is to be operated at altitude in the chamber, then the exhauster capacity must at least equal the total air input, and to meet this requirement with large systems at high altitudes, very extensive plant facilities have to be used. The position can, however, be considerably improved if the ram air system is modified so that the greater part of the air is recirculated by a fan through a second (rig) cooler to extract the heat gained from the system. Such a regenerative installation is illustrated in Fig. 16. The remotely-controlled bleed valve enables the correct ram pressure, relative to chamber ambient conditions, to be maintained and the by-pass valve is included to help bring the ram air up to working temperature from cold at the start of a run, and also to form a useful temperature trimming control throughout the test.

5.1. CHAMBER HEAT BALANCE

Another self-evident, but often forgotten, factor when investigating chamber requirements is the need to consider not only the heat balance of the aircraft system, but also the overall heat balance of the complete chamber installation. To clarify this point it is useful to look at an example taken from an actual simulated flight test, conducted on a fighter installation.

During the run in question, conditions had to be stabilised equivalent to an air speed of Mach 1·03 at 10,000 ft. altitude. To simulate realistic bay temperatures, where the equipment would be installed within the aircraft, the chamber ambient temperature had to be kept at approximately 75°C. At this temperature the heat loss through the lagged chamber walls was small, and since the total excess energy put into the chamber space was approximately 1,720 C.H.U./min. and the outside (ventilating) air temperature was 19°C, some 128 lb./min. of air had to be induced through the chamber for ventilating purposes in order to maintain stable thermal conditions. During this particular test, the total mass flow which had to be dealt with by the exhausters was 241 lb./min. made up as follows: —

Ram outlet flow	81 lb./min.
Cabin outlet flow	32 lb./min.
Ventilating flow	128 lb./min.

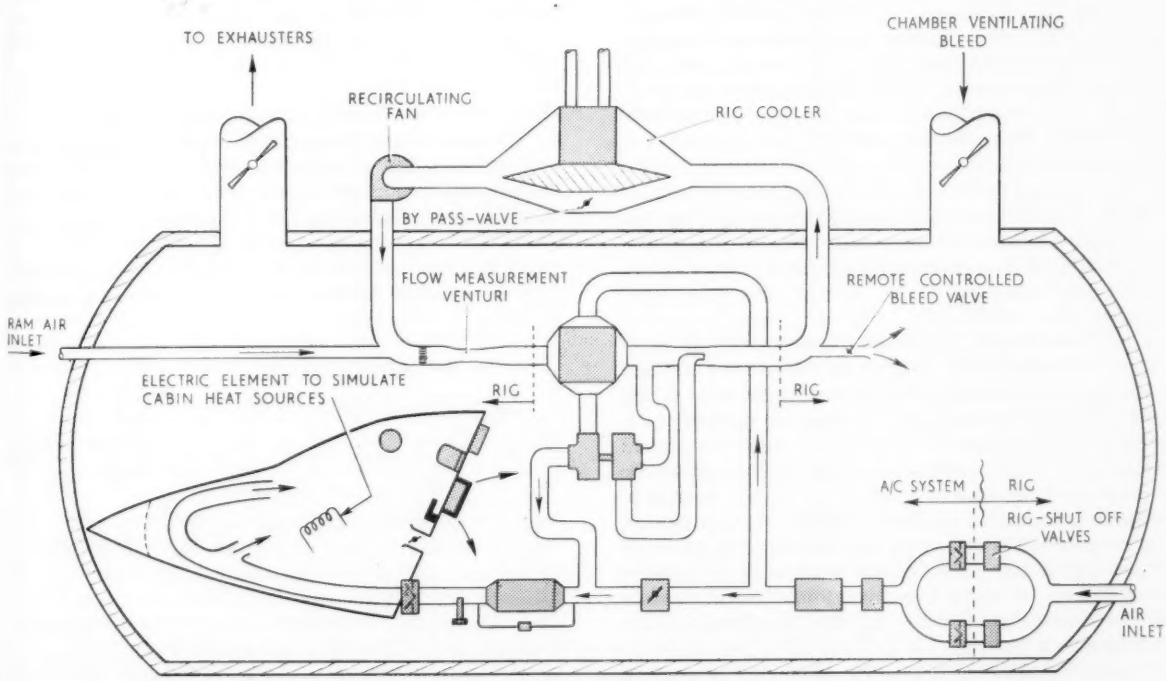


FIGURE 16. Diagram of installation with regenerative ram air supply.

As this total was well within the plant capacity it was of little consequence during the particular run.

Under high altitude conditions, however, and when testing large systems, exhauster capacity is always at a premium and everything possible has to be done to plan the installation with a view to cutting the exhauster air flows to a minimum.

5.2. TYPICAL METHODS OF CUTTING DOWN EXHAUSTER REQUIREMENTS

By ducting the greater part of the ram flow to the inlet of the chamber exhauster pipeline, it is possible to cut down the required ventilating flow considerably while still maintaining the desired ambient conditions around the system. This can only be done at the expense of raising the temperature of the air to the vacuum pumps.

It is also interesting to consider further how the conditions at the exhausters would have been affected had the regenerative ram system using an air to liquid rig cooler (Fig. 16) been adopted, and assuming that only sufficient ram flow is discharged into the chamber to maintain the correct bay temperature conditions. In the test run mentioned the flow discharged from the cabin was 32 lb./min. and the outlet temperature was 28·5°C. The temperature of the ram flow discharged into the chamber was 182°C; thus a flow of some 14 lb./min. would be necessary to maintain the desired 75°C ambient temperature.

This simple estimate does not, of course, take into account details of the ram flow system heat balance which might have been altered somewhat with the change in design. For instance, the re-circulation fan motor (if mounted in the chamber) would dissipate further heat. Dependent on the actual numerical quantities involved, such factors could, in turn, either decrease or increase the mass flows that have to be exhausted from the chamber. These further considerations are, however, only of secondary importance, and do not alter the fact that by simply altering the layout of the ram flow system it is shown that the overall through flow of air and thus the exhauster capacity required, is reduced to about 19 per cent of the quantity necessary for the installation as originally envisaged.

5.3. METHODS OF SIMULATING DIFFERENCES IN TEMPERATURE AROUND SECTIONS OF A SYSTEM

During the test which has just been discussed it was only necessary to regulate the chamber ambient temperature to a mean value of 75°C to achieve the desired conditions. However, in many cases this would not be correct and it is often desirable that various sections of the system should be operated at different temperatures, *e.g.* components which in the aircraft would be mounted in the engine bay frequently have to work in ambient temperatures far higher than those found in other regions of the aircraft. It is thus necessary to simulate these conditions on the rig.

In some cases it is possible to raise the ambient temperature around a section of the system by simply building a shroud over it and lagging this to suit. The radiation and convection from the components themselves being sufficient to alter ambient conditions to the required value. In other cases different desired local temperature variations can be achieved by directing some of either the hot ram discharge air or the relatively cool cabin discharge air over portions of the system. These methods, however, suffer from the fact that they are "hit or miss" in operation and are relatively difficult to adjust once an altitude test is in progress.

The most accurate and easily adjustable method of raising temperature locally above the general ambient level is of course to employ either infra-red lamps, black heat elements or heating tapes, whichever are the most suitable for the particular application. These means, when used in conjunction with a variable transformer, allow quick, easy and accurate adjustment which can be carried out either manually or automatically to a pre-determined programme.

During some tests where a section of the system has to operate at ambient temperatures well below the rest of the installation, it is often more economical in both vacuum capacity and electric power, to maintain the chamber ambient at the higher temperature and to resort to local cooling over the components involved. It is usually best in these circumstances to use a closed cycle cooling circuit re-circulating the cooling air through a heat exchanger and over the components involved by a low power fan, in a similar manner to the ram circuit previously described. Under very high temperature chamber ambient conditions, it is generally safer and more convenient to use a secondary brine cooling circuit rather than to pipe the refrigerant vapour direct to an evaporator mounted in the chamber. Such a brine supply is piped up permanently for general cooling purposes.

5.4. TESTS INCLUDING CABIN COMPRESSORS

In cases where the cabin air supply is tapped from the main engine compressors it is usually satisfactory during rig testing to employ commercial compressors and heaters to supply the air to the system. Sometimes, however, when separate cabin compressors are to be used, it is desirable to test these as part of the rig system, and in this case there are often advantages in mounting

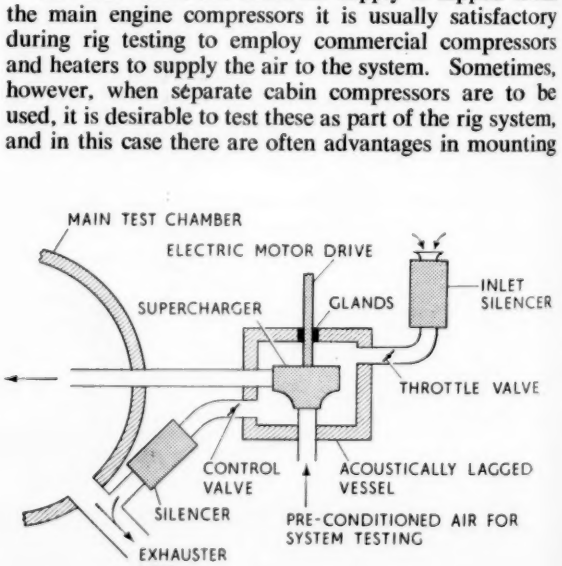


FIGURE 17. Typical cabin supercharger test installation.

them in a small acoustically lagged vessel installed alongside the main test chamber and connected to its exhauster line by a bleed pipe, silencer and control valve. The drives to the compressors are taken from the necessary gear boxes through glands in the wall of the small chamber. The required ventilating air flow is sucked through this vessel via an inlet silencer and throttle valve. The pre-conditioned air required for the system is ducted separately to the blower and then from it to the system in the large chamber. This arrangement, which is illustrated in Fig. 17, allows the blowers to operate under the correct ambient pressures and temperatures and prevents unnecessary heat being dissipated into the main chamber. It also creates more tolerable conditions for observers who may have to work in the vicinity when the system is running.

5.5. TESTS TO CHECK RESPONSE BEHAVIOUR

In practice it is usually found that a great deal of development effort is expended in investigating the ability of the control equipment to catch transient surges with the absolute minimum of disturbance to required conditions. In order that this work can be undertaken great care has to be taken in designing the decompression chamber controls, so that the aircraft performance and cabin flow changes can be realistically simulated. While during most test conditions it is satisfactory for the chamber to be manually controlled occasions frequently arise when exploring transient conditions, when it is of advantage to be able to control the rig automatically, so that similar changes can be repeated accurately during subsequent runs.

5.6. CONSIDERATION OF CABIN VOLUME

One of the most important factors affecting the time constant and general behaviour of environmental control installations is cabin volume and, so far as is possible, it is best to test the systems in cabins of representative size. In the case of military aircraft this usually raises few problems, but this is not so with equipment for big civil machines where very large volumes are involved.

On the chamber described, steps have been taken to connect up the airlock via a 3 ft. diameter low resistance pipe and valves to two other nearby chambers so that by mounting the pressure control equipment on adaptors leading into the airlock a cabin volume of 7,400 cubic ft. can be simulated discharging into an ambient space of 7,000 cubic ft.

When simulating aircraft ground pressure test conditions the total volume can be used, giving a cabin up to 14,400 cubic ft.

5.7. CONSIDERATION OF CHAMBER (AMBIENT) VOLUME

During flight the ambient air pressures are unaffected by the discharge of air from the cabin, but under chamber test conditions, however, the ambient pressure tends to vary directly as the flow is changed and corrective control action has to be taken to counteract this. The larger the ambient volume relative to the "through flows" involved the slower the rate of pressure change, and thus the easier it is to hold the desired pressure. At first sight, therefore, it appears that "the larger the better" but, in practice, the ambient volume is limited by the ability of the exhausters to extract air and maintain the desired maximum rate of pressure change (aircraft climb). For reference, Fig. 18 shows the additional weight of air that has to be extracted from a vessel at various altitudes and temperatures for each 1,000 ft./min. rate of climb per 1,000 cubic ft. of volume.

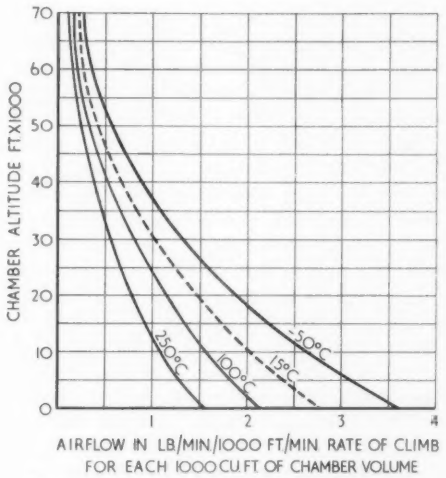


FIGURE 18. Exhausted air flow/climb curves.

6. General Discussion

A number of decompression chambers have been installed for experimental purposes by firms and research organisations throughout this country. Few of these, however, are backed by sufficient plant capacity to allow full scale environmental control system tests to be carried out on complete aircraft installations.

The chamber described was designed as an integral part of the equipment in a large high altitude laboratory so that when necessary the full plant resources can be tapped. It is therefore possible to obtain high air flows when the occasion demands while still maintaining a much higher plant utilisation rate than is usual with test installations of this nature.

As originally designed the facility was intended principally for conducting high temperature tests and apart from the coolers in the ram air supply line and for cooling equipment locally, no provision was made for lowering the general internal working temperature below ambient conditions. Experience has shown, however, that additional low temperature work could also be usefully carried out. To meet this need a design has been prepared in conjunction with J. and E. Halls Ltd. for a removable evaporator system with re-circulating fans which could be wheeled into the chamber for cold runs and taken out again when not required. It has been calculated that this installation will lower the internal temperature to $-45°C$ when connected to an extension line from existing laboratory refrigeration plant.

In common with most other items of research equipment, the uses to which the facility is actually being

put go beyond those for which it was originally designed. As a typical example of this, during a recent test programme, heavy airborne equipment mounted on a specially designed suspension was successfully vibrated by means of two synchronised 3,000 lb. thrust vibration tables while operating at high altitudes. Neither the structure nor the foundations of the chamber were designed with this type of work in view, but in the event no trouble of any kind was experienced.

In the past the performance of most aircraft has been such that it has been possible to conduct a great deal of development flying before it has been necessary to bring the environmental control installations into full operation. This fact, coupled with the general lack of suitable test facilities and a tendency for some designers to underestimate the problems, has led to many aircraft installations remaining virtually untested until actually being required in flight. The inevitable result has been that often inadequate and unreliable systems have been produced and expensive modifications have had to be introduced at a late stage in the development of the aircraft.

The position has been further complicated in many cases by the fact that systems have been designed utilising components made by a number of specialist sub-contractors and due to lack of realistic specifications these have not been in a position to assess fully the part their units are called upon to play in the overall installations.

As the operating speeds and altitudes of machines increase it is becoming more important that full cabin and equipment bay air conditioning should be obtained at an early stage of aircraft flight development. This, together with experience of past difficulties, is leading to a far greater appreciation of the advantages to be gained by supplementing project calculations and independent component development with a properly planned decompression chamber system test programme. Ideally this work should be undertaken before flight trials begin. In fact the earlier it can be completed the better, as modifications to aircraft systems are frequently found necessary and it is a great advantage if these can be incorporated from the start in prototype as well as in production aircraft.

During the past few years considerable experience and practical "know how" has been gained in the high altitude laboratory on the problems associated with the rigging, instrumentation and testing of complete installations; a great deal more could undoubtedly be written without exhausting the subject. However, it is hoped that sufficient general information has been given to illustrate many of the most important factors involved and to show that the decompression chamber constitutes a useful facility when developing aircraft environmental control installations and for other work involving tests covering a wide envelope of air flow altitude and temperature conditions.

Acknowledgments

The author wishes to acknowledge his gratitude to Dr. E. W. Still, F.R.Ae.S., and the Directors of Normalair Ltd. for permission to publish this paper. He would also like to thank Mr. W. W. Baker, Mr. R. J. Furneaux and other colleagues for information, constructive comments and help with the diagrams during the course of its preparation. The views expressed are personal and do not necessarily represent those of the Company.

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1. Authority: British Civil Airworthiness Requirements.
2. Tables of Temperature Relative Humidity and Precipitation for the World.

TECHNICAL NOTES

A comment by Professor Argyris and Mr. Kelsey on Dr. Grzedzielski's original note ("Some Applications of the Matrix Force Method of Structural Analysis, p. 354, June 1960 JOURNAL) was published in the August 1960 JOURNAL. Dr. Grzedzielski sent a reply to which Professor Argyris and Mr. Kelsey replied at length. As both sides obviously held strong views these "contributions" were held over so that with their final comments—so far as this section of the JOURNAL is concerned—they could be published in one issue.—Ed.

The Initial Strain Concept

ALEX. L. M. GRZEDZIELSKI, Dr.Eng., P.Eng.
(National Aeronautical Establishment, Ottawa)

IN A NOTE published in the June 1960 JOURNAL[7], I pointed out certain deficiencies in the initial strain concept introduced by Messrs. J. H. Arygris and S. Kelsey[1]. In the August JOURNAL[8], those authors have disputed my conclusions with an argument which, I feel, does not pay due regard to current literature and I believe may be based on a misinterpretation of certain algebraic manipulations contained in my note. For they concluded that my note contained misprints, which it, in fact, did not. Due to these unfortunate circumstances Messrs. Argyris and Kelsey failed to illuminate the basic disagreement between our two formulations. For this reason I should like to set out in simple detail the exact nature of our disagreement and to point out, as clearly as possible, where the initial strain concept as developed by Messrs. Argyris and Kelsey becomes, I believe, somewhat inadequate and how the concept of fictitious thermal loads achieves greater validity.

NOTATION

A	lumped flange area
D	matrix $b_1'fb_1$
E	Young's modulus of elasticity
G	applied load in terms of shear flow
H	generalised strain
I	unit matrix
L	length as shown in Fig. 1
R	applied load
S	stress
θ	temperature
b	matrix, relating stress to the applied load
b_0	matrix, relating statically determined stress to the applied load
b_1	matrix, relating stress to the redundant load
f	flexibility matrix
f_h	submatrix of f of the order $h\times h$
f_h^*	submatrix of f of the order $(g+h)\times h$
f_{mh}	submatrix of f of the order $h\times h$ after modification
f_{ah}	flexibility matrix of the additional material
k	diagonal matrix of cross-sectional changes of Ref. 7
t	thickness
α	coefficient of thermal expansion
ν	Poisson's ratio
h	index denoting the modified elements
g	index denoting the elements which remain unchanged

THERMAL STRESS ANALYSIS

The concept of fictitious thermal loads is proved to all satisfaction in Refs. 4, 5 and 6. In particular, Ref. 6 asserts that "because of the equivalence of the thermal stress to a fictitious body force, all . . . elastic results (reciprocal theorem, ... general solutions, variational methods, and so on) can be applied here also." This opens the way of using the Castigliano theorem of least work in connection with the fictitious thermal loads and Matrix Force Method. The following is an example of the matrix operations required.

Let Fig. 1 represent a portion of an upper panel of a multi-spar and multi-rib wing shown in Fig. 2 of Ref. 1. Let the full lines represent the lumped flanges of spars and ribs so that the connecting sheet of thickness t is required to carry shearing stresses only. It is assumed, however, that the flexibility matrix f contains several coupling terms accounting for the Poisson's ratio of the material. These are discussed in Ref. 3. For convenience it is assumed also that the lumped flange area is $A=Lt$. We propose to find stresses due to the temperature distribution defined by means of four values at the four corners of the panel.

In the idealised structure the following stress values are considered:

(*a*) spar flange direct stress at cross sections 1, 2, 3, 4,
(*b*) rib flange direct stress at cross sections 5, 6, 7, 8,
(*c*) panel shearing stress represented by one value only.

These stresses comprise a column S. Assuming that the matrix $b_1D^{-1}b_1'$ for the whole structure has been computed and is available, the thermal stress at the above nine locations is given by the formulae:

Equation (49), Ref. 1

$$S=-b_1D^{-1}b_1'H \quad . \quad . \quad (1)$$

Equation [5], Ref. 7

$$S=-b_1D^{-1}b_1'fb_{0\theta}\theta. \quad . \quad . \quad (2)$$

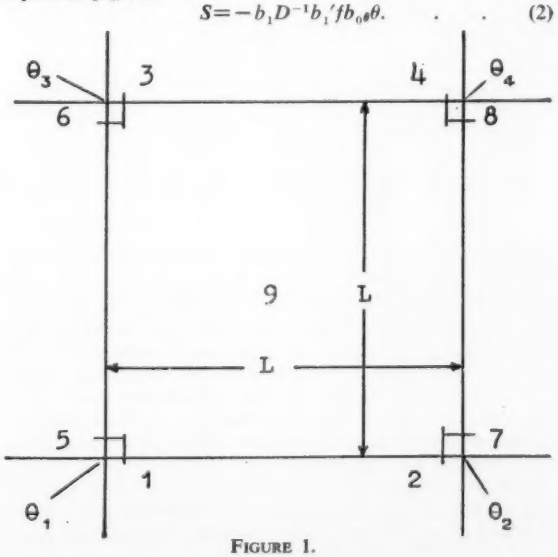

FIGURE 1.

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In the following only the differences in the entries H and $fb_{0\theta}\theta$ are discussed. It should be understood that finally and as indicated above, the structure under consideration is to be the whole wing and not just the panel shown in the drawing. For the same reason the columns H of (3) and (4) are only a small portion of the total input.

The entry of the equation (1) is given by equation (25) of Ref. 1. It is

$$\begin{vmatrix} H_1 \\ H_2 \\ H_3 \\ H_4 \\ H_5 \\ H_6 \\ H_7 \\ H_8 \\ H_9 \end{vmatrix} = \alpha L/6 \begin{vmatrix} 2 & 1 & . & . \\ 1 & 2 & . & . \\ . & . & 2 & 1 \\ . & . & 1 & 2 \\ 2 & . & 1 & . \\ 1 & . & 2 & . \\ . & 2 & . & 1 \\ . & 1 & . & 2 \\ * & * & * & * \end{vmatrix} \times \begin{vmatrix} \theta_1 \\ \theta_2 \\ \theta_3 \\ \theta_4 \end{vmatrix} \quad . \qquad (3)$$

where all elements indicated by . and * are zero for a non-swept wing. The entry of the equation (2) is of the same dimensions as the one of (3) and is denoted by the same letter H. Then

$$\begin{vmatrix} H_1 \\ H_2 \\ H_3 \\ H_4 \\ H_5 \\ H_6 \\ H_7 \\ H_8 \\ H_9 \end{vmatrix} = \begin{vmatrix} & & & & & & & & \\ & & & & & & & & \\ & & & & & & & & \\ & & L & & & & & & \\ & & 6AE & & & & & & \\ & & & & & & & & \\ & & & & & & & & \\ & & & & & & & & \\ & & 1/Gt & & & & & & \end{vmatrix} \begin{vmatrix} 2 & 1 & . & . & -3\nu & . & . & . & * \\ 1 & 2 & . & . & . & . & -3\nu & . & * \\ . & . & 2 & 1 & . & -3\nu & . & . & * \\ . & . & 1 & 2 & . & . & . & -3\nu & * \\ -3\nu & . & . & . & 2 & 1 & . & . & * \\ . & . & -3\nu & . & 1 & 2 & . & . & * \\ . & -3\nu & . & . & . & . & 2 & 1 & * \\ . & . & . & -3\nu & . & . & 1 & 2 & * \\ * & * & * & * & * & * & * & * & 1 \end{vmatrix} \times \frac{\alpha AE}{1-\nu} \begin{vmatrix} 1 & . & . & . \\ . & 1 & . & . \\ . & . & 1 & . \\ . & . & . & 1 \\ 1 & . & . & . \\ . & . & 1 & . \\ . & 1 & . & . \\ . & . & . & 1 \\ * & * & * & * \end{vmatrix} \times \begin{vmatrix} \theta_1 \\ \theta_2 \\ \theta_3 \\ \theta_4 \end{vmatrix} \quad . \qquad (4)$$

where the 9×9 matrix is the flexibility matrix in terms of forces and includes the Poisson's ratio coupling. The 9×4 matrix is the $b_{0\theta}$. It expresses the statically indeterminate part of the final stress (*i.e.* due to the fictitious thermal loads), at the considered cross sections and is computed from equation (1) of Ref. 7. Multiplying the matrices out, one obtains an entry which is not the same as that given by (3)

$$\begin{vmatrix} H_1 \\ H_2 \\ H_3 \\ H_4 \\ H_5 \\ H_6 \\ H_7 \\ H_8 \\ H_9 \end{vmatrix} = \begin{vmatrix} & & & & & & & & \\ & & & & & & & & \\ & & & & & & & & \\ & & \alpha L & & & & & & \\ & & 6(1-\nu) & & & & & & \\ & & & & & & & & \\ & & & & & & & & \\ & & & & & & & & \\ & & & & & & & & \end{vmatrix} \begin{vmatrix} 2-3\nu & 1 & * & * \\ 1 & 2-3\nu & * & * \\ * & * & 2-3\nu & 1 \\ * & * & 1 & 2-3\nu \\ 2-3\nu & * & 1 & * \\ 1 & * & 2-3\nu & * \\ * & 2-3\nu & * & 1 \\ * & 1 & * & 2-3\nu \\ * & * & * & * \end{vmatrix} \times \begin{vmatrix} \theta_1 \\ \theta_2 \\ \theta_3 \\ \theta_4 \end{vmatrix} \qquad (5)$$

Comparing (3) and (5) one can see clearly that the results of Ref. 1 and of Ref. 7 differ due to the Poisson's ratio coupling which is not negligible in multi-spar and multi-rib wing structures. However, when the structural elements do not influence each other so that the terms containing the Poisson's ratio do not appear, in other words when the flexibility matrix is diagonal, at least in the sense of equation (33) of Ref. 1, the results are exactly the same.

When the wing is swept, the coupling between structural elements is more involved. In this case all matrix elements indicated by * are expected to be filled out. In equation (3), if I am not mistaken, the 9th row of the 9×4 matrix will be called upon to express the panel contribution as a function of the average thermal strain. On the other hand, in equation (4) the 9th row of the matrix 9×4 will express the statically determinate portion of the thermal stress while the 9th row and the 9th column of the 9×9 matrix will express the swept wing coupling.

From the foregoing it seems clear that the concept of fictitious thermal loads must lead to a more acceptable result than does the concept of initial strain.

CUT-OUT PROCEDURE

In Section 4 of their reply Messrs. Argyris and Kelsey present a lengthy argument which can be reduced to the following three objections as to my point of view:

(*a*) Terms b_1' in equation [9], [10], [11], [17], [18] of Ref. 7 should be appended by an index h and written b_{1h}'.

(*b*) The right hand term of the second equation [17] is incorrect.

(*c*) The two equations [18] lead to a single equation (71) of Ref. 1 which is only a special case of the equation (68) of the same reference.

I cannot agree with the above statements. As the quantity G of Ref. 7 is a load, the stress due to this load is given by the known formula:

$$S=(b_{0G}-b_1D^{-1}b_1'fb_{0G})\,G \quad . \quad . \qquad (6)$$

Now, for this particular loading b_{0G} is of the order $(g+h)\times h$. It is composed of two submatrices: one, of the order $g\times h$, is a zero matrix, the other, of the order $h\times h$, is a unit matrix. Hence the matrix fb_{0G} is of the order $(g+h)\times h$ and consists of only a few columns of f. In Ref. 7 this matrix has been denoted by the symbol f_h and the fact that f_h is not a square matrix was indicated by omitting the index h in the pre-multiplying matrix b_1'. The assumption is that the index h modifies the matrix f in the same manner as it modifies the matrix b_1, namely by suppressing all rows or columns which do not belong to the range h. In connection with the present text, however, and to avoid all possibility of misunderstanding, the notation $fb_{0G}=f_h{}^*$ is preferable. Then the above stress formula, partitioned into the g and h ranges, becomes

$$\left.\begin{aligned} S_g &= -b_{1g}D^{-1}b_1'f_h{}^*G \\ S_h &= (I_h-b_{1h}D^{-1}b_1'f_h{}^*)\,G \end{aligned}\right\} \quad . \quad . \qquad (7)$$

Evidently the notation $b_{1h}'f_h$ as suggested by Messrs. Argyris and Kelsey would be correct but only if there were no terms in the field $g\times h$ of the matrix $f_h{}^*$. However, my note[7] was written precisely because there can be terms in this matrix field. This answers objection (*a*).

Objection (*b*) can also be answered quite easily. After a term b_hG is added to the second equation (7), one can ascertain at once that because of [17] of Ref. 7 the second equation [18] is correct as written.

To decide whether the two controversial derivations lead to the same result, equation [17] of Ref. 7 is compared with equation (66) of Ref. 1. In doing so, however, it is convenient to have the modifications to the flexibility matrix defined in the same manner. Thus condition [17]

is written in terms of flexibilities defined in Ref. 1

$$-f_{ah}G=f_h\,[b_hR+(I_h-b_1D^{-1}b_1'f_h^*)\,G] \quad . \qquad (8)$$

where $f_{ah}^{-1}=f_{hm}^{-1}-f_h^{-1}$ is the stiffness matrix of the additional material. (It should be pointed out here that the matrix k of my previous note was unduly restricted to a diagonal form. The conversion formula is: $k=f_m^{-1}f_h-I$.) After manipulation (8) becomes

$$f_hG=(f_{hm}-f_h)\,[b_hR-b_{1h}D^{-1}b_1'f_h^*G] \quad . \qquad (9)$$

and (66) becomes

$$H=(f_{hm}-f_h)\,[b_hR-b_{1h}D^{-1}b_{1h}'H] \quad . \qquad (10)$$

It appears that equations (9) and (10), together with (68) of Ref. 1 and [18] of Ref. 7 may lead to the same result if and only if the transformation

$$H=f_hG \quad . \quad . \quad . \qquad (11)$$

is possible. But this depends on whether the field $g\times h$ of f_h^* is free from coupling elements or not. This answers objection (*c*).

To complete the argument it may suffice now to give some examples of coupling between structural elements such that terms are introduced in the above matrix field. Suppose that a structural element of the range h is a spar flange made of a portion of the covering sheet and a few stringers (lumping) and that the corresponding element of the range g is a (lumped) flange running crosswise. As coupling occurs in the sheet only, it is clear that any modification of the stringer size of the element h does not affect either the flexibility of the cross flange or the numerical value of the coupling term. Similarly, if the structural element of the range g is the sheet itself, considered as a shear carrying element, the numerical value of the matrix element expressing swept wing coupling will also remain unchanged. Other examples of intricate coupling can be found in cases when precalculated flexibility submatrices of frames, bulkheads and so on are inserted in the general flexibility matrix. In all these cases severing of a connection does not affect the flexibility matrix of a subassembly. It should be noted at this place, that when a change occurs in the thickness of the covering sheet, then all structural elements made of this sheet belong to the range h and define a matrix f_{ah}.

Having arrived at the conclusion that the formulations of Ref. 1 and Ref. 7 are not equivalent in the general case, when intricate coupling exists, it remains to decide which of the derivations is more likely correct. As I see it, the concept of initial strains as formulated by Messrs. Argyris and Kelsey lacks appeal because it can only yield correct results when the state of strain of structural elements is one-dimensional. Thus I would suggest that their theory is incomplete in the present formulation because it deals with only one component of the strain tensor at a time, in other words because it has not been presented, as yet, in a general and invariant way. On the other hand, as has long been recognised, there exists a true equivalent between thermal stresses and fictitious body forces, to the analysis of which the proven theorems of the elastic theory may be rigorously applied. It is therefore to be expected that, as matters stand, the derivations which I have proposed may have the greater validity.

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The Validity of the Initial Strain Concept

J. H. ARGYRIS and S. KELSEY

(*Department of Aeronautics, Imperial College of Science and Technology*)

1. INTRODUCTION

We must thank Dr. Grzedzielski for the clarification of his argument and notation contained in his second note[2]. Unfortunately, this note only shows more clearly the fundamental misconceptions and errors in his arguments. Our own argument as to the generality and validity of the initial strain concept remains unaffected.

In our previous note, we were concerned to explain the initial strain concept, which Dr. Grzedzielski had criticised as "an oversimplification not generally admissible" both in the calculation of thermal stresses and as a device to simulate the effects of structural cut-outs and modifications. Here, we prove independently by a direct consideration of the compatibility conditions in the final structure, that the modified stress distribution given by our analysis is the correct one—regardless of the extent of the coupling in the flexibility matrix.

We also show that our approach to the thermal stress problem, when applied to a particular example, yields results in accordance with the predictions of the exact Theory of Elasticity: Dr. Grzedzielski's analysis does not.

It is unfortunate that the argument by which the meaning of the symbol f_h in Ref. 1 was supposed to be deduced, should be quite outweighed by Dr. Grzedzielski's

explicit statement that our own symbols and notation were used wherever possible. As mentioned above, the presence of coupling terms in the flexibility matrix does not affect the analyses by initial strains. But it does affect his analysis of the modifications—more than he seems to be aware. In fact, as presented in Ref. 1 his analysis only makes sense if there is no coupling, and in this case the result is identical with ours. In the more general case, for which he claims the greater validity, his analysis contains serious errors. Correction of these errors and a closer examination of the question than he has given it show that the different result is due to the basic difference of definition of the modification from that which we have used. For the cut-out, his analysis is proved here to be definitely wrong with coupling terms in the flexibility matrix.

It is unreasonable to claim greater validity because the analyses of Refs. 1 and 2 are based on the concept of fictitious thermal loads " as proved to all satisfaction " and " to the analysis of which, the proven theorems of elastic theory may be rigorously applied "[2]. The " proof to all satisfaction " applies to exact solutions of the Theory of Elasticity and for these, all principles are equally valid which rest on the fundamental ideas of equilibrium of stresses and compatibility of total strains.

The question at issue is the application of the principles to idealised systems. The validity of the principle itself in a non-idealised system has no bearing on this. Our view is that in the analysis of redundant systems by forces, the redundancy equations follow from the compatibility conditions of *total* strains. In this respect we treat the initial (or thermal) strains in exactly the same way as the elastic strains. The Castigliano Theorem of Least Work, in its conventional form, deals only with elastic strains due to stress. By his insistance on using this form of the compatibility condition, Dr. Grzedzielski is, in fact, forced to use the concept of fictitious thermal loads. As we stated before, their only purpose is to pretend that the thermal strains are elastic ones due to load. Were he to use a more modern version of the Castigliano Theorem[6], including thermal effects, he would in fact derive results—though by a more involved analysis—in agreement with ours.

The concept of fictitious thermal loads is, in any case, completely irrelevant to Dr. Grzedzielski's argument in the cut-out and modification analysis. In his first note[1] he explicity states that his " reasoning replaces that of (his) Ref. 1 without resort to thermo-elastic concepts "—because he objected to demons blowing hot or cold on the structure. His approach to these questions is, in fact, fundamentally the perturbation method initiated by Van der Neut and Cicala and made more generally applicable by using the matrix analysis.

In the following Sections and Appendices, we present the detailed discussion of the three main topics.

2. THERMAL STRESS ANALYSIS

Apart from his method of approach, the most notable feature of Dr. Grzedzielski's example[2] is the existence of Poisson's ratio ν in the quantities **H**. From a physical argument it is difficult to see how this can be logical. As calculated by him, **H** represents the generalised strains caused by the fictitious thermal loads. But the fictitious thermal loads are those required to get rid of the initial stresses associated with complete suppression of the thermal expansion. The stresses due to the thermal loads applied on the statically determinate basic system are therefore exactly equal and opposite to the initial stresses and the basic system must as a result be completely free of stress. In fact, this is implied in Dr. Grzedzielski's analysis because, as he says, it is unnecessary to calculate the actual fictitious loads, but only the stresses due to them And these are obtained simply by putting a negative sign to the initial stresses.

The quantities **H**, therefore, can only be the direct consequence of the thermal expansion in the statically determinate system. As such, they can be calculated directly by the same procedure as used for the elastic strains. This is the method we have adopted. There is no unwarranted simplification involved.

Since **H** is due only to thermal expansion of the basic system, it should not involve ν if it has been calculated consistently. The fact that ν is still present in Dr. Grzedzielski's expression suggests that the thermal loads do not completely eliminate the initial stresses and that the calculation of **H** is somewhere inconsistent.

The example given by Dr. Grzedzielski is essentially that of the thermal stresses in a flat isotropic sheet, as calculated by the Matrix Force Method. For such a sheet, the Theory of Elasticity shows that the stress distribution due to applied loads is independent of the elastic properties of the sheet, provided there are no boundary conditions of the kinematic type (*i.e.* no restrictions on the displacements) and the body forces are constant (Ref. 10, p. 25). What interests us here is that the stress distribution in such a case is independent of Poisson's ratio and this independence of ν holds also for the stress distribution due to non-uniform temperature* (Ref. 6, p. 14). We may therefore test the consistency of the Matrix Force Method, by using it to analyse a typical case where exact theory predicts that the final stress distribution is unaffected by Poisson's ratio.

A simple problem of this kind is investigated in Appendix A. It is shown there that Poisson's ratio disappears from the analysis in precisely the circumstances described above, when the stresses are caused by applied forces. The overall consistency of the idealisation and, in particular, of the flexibility matrix is thus satisfyingly confirmed. This is true also of the thermal stress analysis in the method given by us, which Dr. Grzedzielski has correctly interpreted in his second note.

On the other hand, the method of fictitious thermal loads, as applied by Dr. Grzedzielski, gives a stress distribution which depends on Poisson's ratio. Because of the consistency of the remainder of the analysis, this dependency on ν is entirely due to its presence in the expressions he derives for **H**.

In the more general problem of a plate with reinforcing flanges, such as the surface of a multi-spar, multi-web wing, Poisson's ratio will, of course, no longer disappear from the analysis in the above way. But little confidence can be placed in the calculation of its effect by Dr. Grzedzielski's method, when this method is shown to introduce ν in a problem where it should have no effect.

The only argument advanced by Dr. Grzedzielski for the greater validity which he claims for his analysis is that it is based on the method of fictitious thermal loads. But

*This is easily shown by forming the compatibility conditions on the total strains (*i.e.* thermal plus elastic) in terms of the Airey stress function F. We find

$$\Delta^2 F + \alpha E \Delta\theta = 0$$

where $$\Delta = \frac{\partial^2}{\partial x^2} + \frac{\partial^2}{\partial y^2}$$

is the Laplace operator in two dimensions.

as we have shown in the two preceding sections, this leads to an inconsistency in that it shows the stress distribution to depend on Poisson's ratio in cases where exact theory explicitly denies such a dependence. The reason is that the thermal loads are calculated, from the direct stresses required at the grid intersection points to suppress the thermal strain there, by a straightforward finite difference approximation to the actual structure. But these loads are then applied to a flexibility matrix which is more than a finite difference approximation—as shown by the presence of coupling terms between the two ends of a flange element. In a consistent analysis one would expect the thermal load matrix $\mathbf{b}_{0\theta}$ to display a similar coupling between the flange loads and the temperatures at neighbouring grid points. To derive such a $\mathbf{b}_{0\theta}$ is not, of course, practical and to this extent the inconsistency in the treatment of Poisson's ratio is inherent in the use of the fictitious thermal load method for the particular problem. But this method of attack is dictated solely by Dr. Grzedzielski's desire to force the analysis into the conventional Castigliano pattern of a structure loaded by forces. The fact that it leads in this case to inconsistencies supports again our argument[3] that it is not the natural approach to the problem.

A fundamental difference between Dr. Grzedzielski and ourselves is that he does not base his analysis on an explicitly defined idealised structure. His approach is rather to consider the idealisation as the approximations necessary in evaluating the strain energy integrals required in the actual structure. This is no doubt the reason for his argument that the initial strain concept is an oversimplification and "deals with only one component of the strain tensor at a time." We repeat that the initial strains are *generalised strains* in the *idealised structure*. As such they cannot be discussed as though they were the unit strains in the actual structure. We must again emphasise that the general matrix analysis in our work is for an explicitly idealised structure, and the question of the accuracy or otherwise of the results in a particular case is a consequence of the idealisation, not of the general analysis. Dr. Grzedzielski does not make this sharp distinction, partly because his method of approach does not allow it, and erroneously attributes differences between us to lack of generality in the concepts and analysis of the idealised structure. As we show further, this also accounts in part for his invalid criticism of the cut-out and modification method.

3. ANALYSIS OF CUT-OUTS

The different meaning which Dr. Grzedzielski attached to the symbol $\mathbf{f}_h$ in his original note does, of course, result in a difference between his result and our own for the stresses in the structure with cut-outs. Unfortunately, his result is incorrect when the element to be cut-out is elastically coupled to structural elements which remain. This is demonstrated in Appendix B by an investigation of the compatibility conditions in the final structure, *i.e.* with cut-outs. There, we consider the redundancy equations which would be formed by a direct consideration of the final structure and it is shown that they are exactly satisfied by the stress distribution which we have given[4, 5, 6], but not by the results of Dr. Grzedzielski's analysis. The reason is that, in his method, the element or part element eliminated is still subject to stress when it is removed—under the forces **G**. Because of the coupling, the supposedly eliminated element still affects the strains in the rest of the structure. Conversely it follows that the same coupling leads to strains in the eliminated element which are not those of a free element solely under the forces **G**. Dr. Grzedzielski's cut-out is not complete because the coupling is not destroyed.

The reason why the initial strain approach is still valid with coupling in the flexibility matrix is that the element to be eliminated is reduced to a stress-free condition. The coupling has, therefore, no effect.

In our previous note we presented a physical picture of the process of cut-out simulation, into which both methods of analysis could be fitted. Against this common background we pointed out the reason for the limitation on Dr. Grzedzielski's analysis and why it does not apply in our own case. Our understanding of his argument was that he had appreciated the limitation and had mistakenly assumed the initial strain approach also to be limited in the same way. Unfortunately, Dr. Grzedzielski, in reducing our "lengthy argument" to three points of algebra, has ignored this important point. Perhaps he will find the detailed algebra of Appendix B more convincing.

We have been reproached in Ref. 2, for "a disregard of current literature." We are not quite sure which literature we are guilty of ignoring, but may we here point out that our results for the cut-out—and also for the modification analysis—have been independently confirmed in the published work of Goodey[8, 9]. In both cases, the results follow from formal mathematical arguments, though in the case of the cut-outs the Lagrangian multipliers used by Goodey may be interpreted physically as our initial strains.

4. ANALYSIS OF MODIFICATIONS

It is important first to realise that the modification problem posed and solved in Refs. 4 and 5 is concerned with modifications defined in the idealised structure. Given that the idealised structure has originally an unassembled flexibility matrix $\mathbf{f}$, we determine the change in the stress distribution when the (square) submatrix $\mathbf{f}_h$ in $\mathbf{f}$ is changed to $\mathbf{f}_h+\Delta\mathbf{f}_h$. Athough, for simplicity, the original derivation[4, 5] was presented with $\mathbf{f}_h$ not coupled to $\mathbf{f}_g$, it is easily verified that the result is unaffected by the presence of coupling. In addition to this, and as we said in our previous note, the result is confirmed explicitly[5], without reference to the concept of initial strains and with no limitation of the form of $\mathbf{f}$, by the stress distribution satisfying the compatibility conditions in the modified structure. As in the case of the cut-outs this is proved by substituting the result in the equations which would be set up in a direct analysis of the modified structure. Since our previous mention of this fact appears to have gone unnoticed, we present the complete algebraic demonstration in Appendix C.

It is clear from the previous paragraph and the compatibility proof of Appendix C that the validity of the result in no way depends on the form of the flexibility change $\Delta\mathbf{f}_h$. As far as the analysis of the idealised structure is concerned we could in principle take $\Delta\mathbf{f}_h$ to be quite arbitrary—provided only it does not make singular the matrix to be inverted. The generality of the definition and analysis of the problem thus facilitate its use in such apparently different applications as the correction of errors in parts of $\mathbf{f}$ and allowance for plastic strains as well as the limiting case of a cut-out. This generality is moreover unaffected by any coupling terms in $\mathbf{f}$. It is only necessary that the change $\Delta\mathbf{f}_h$ include all alterations to the flexibility matrix $\mathbf{f}$ of the idealised structure.

The analysis by the perturbation method, however, starts from a different basic definition of the modification —it is defined there as a parallel addition of reinforcement (which may be positive or negative) to certain of the

elements. It is, of course, possible to calculate the resulting change in **f**, as has been attempted by Dr. Grzedzielski and hence in effect to redefine the modification in our terms. But there is an important difference when the reinforced elements are coupled to the rest of the structure. For in this case **f** is changed not only for the reinforced elements but also for all elements to which they are coupled. In fact, it is not difficult to show that the new flexibility $\mathbf{f}_m$, due to the reinforcement, is given by

$$\mathbf{f}_m=\mathbf{f}-\mathbf{f}_h^*\,[\mathbf{f}_h+\mathbf{f}_{ah}]^{-1}\,\mathbf{f}_h^{*\prime} \quad . \quad . \quad (1)$$

Dr. Grzedzielski's analysis of Ref 2 is, in this respect, incorrect since he here ignores the effect of coupling in calculating his version of equation (1) (line following equation (8) of Ref. 2). As a special case he also states incorrectly that, in a reinforced sheet, a change to the actual flange area does not affect either the flexibility of the cross flange or the magnitude of the coupling term due to the Poisson's ratio of the sheet. His argument, that the coupling only occurs through the sheet and is hence unaffected, is only valid for coupling expressed in a stiffness matrix. When the stiffness is inverted to form the flexibility, the coupling does enter and results in a transformation of the form of equation (1). This result is also clear physically. For example, the restriction of transverse strain in a sheet, by the addition of transverse stiffeners, results in a decreased longitudinal flexibility. In the limit, when the transverse stiffeners are made rigid and the cross strains completely suppressed, we obtain the well known result that the effective longitudinal E becomes $E/(1-\nu^2)$.

We see therefore, that the comparison made by Dr. Grzedzielski in his second note[2] between equations (9) and (10) is not valid, because the matrices

$$\Delta\mathbf{f}_h=\mathbf{f}_{hm}-\mathbf{f}_h$$

do not mean the same in the two equations when the modified elements are coupled to the rest of the structure.

Apart from the above question of interpretation, the perturbation analysis presented in Ref. 2 is in fact incorrect, because it fails to take the coupling into account consistently and it also omits to incorporate the reinforcement into the structure in the final stress distribution. Because of the coupling, the strains $\mathbf{v}_h$ in the h-elements, caused by application of the perturbation forces **G**, depend on the stress distribution in the whole structure—not simply the h-elements, as assumed by Dr. Grzedzielski. The result is:

$$\mathbf{v}_h=\mathbf{f}_h^{*\prime}\,[\mathbf{bR}-\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_1{}'\mathbf{f}_h^*\mathbf{G}]+\mathbf{f}_h\mathbf{G} \quad . \quad (2)$$

Equating these to the strains caused in the reinforcement itself by the forces $(-\mathbf{G})$ applied to it, we find, instead of equation (8) of Ref. 2,

$$-\mathbf{f}_{ah}\mathbf{G}=\mathbf{f}_h^{*\prime}\mathbf{bR}-[\mathbf{f}_h^{*\prime}\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_1{}'\mathbf{f}_h^*-\mathbf{f}_h]\,\mathbf{G} \quad . \quad (3)$$

and therefore

$$\mathbf{f}_h^{*\prime}\mathbf{bR}=[\mathbf{f}_h^{*\prime}\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_1{}'\mathbf{f}_h^*-(\mathbf{f}_h+\mathbf{f}_{ah})]\,\mathbf{G} \quad . \quad (4)$$

Now although equations (7) in Ref. 2 are correct for the stresses in the original structure due to **G**, these are not what we require. The stresses in the final structure must include those in the reinforcement and hence a term $(-\mathbf{G})$ has to be added to the second of equations (7)[2]. These two equations now combine to yield the final stress distribution in the modified structure

$$\mathbf{S}_m=\mathbf{bR}-\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_1{}'\mathbf{f}_h^*\mathbf{G} \quad . \quad . \quad (5)$$

or, with substitution for **G** from (4)

$$\mathbf{S}_m=\mathbf{bR}-\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_1{}'\mathbf{f}_h^*\,[\mathbf{f}_h^{*\prime}\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_1{}'\mathbf{f}_h^*-(\mathbf{f}_h+\mathbf{f}_{ah})]^{-1}\mathbf{f}_h^{*\prime}\mathbf{bR} \quad . \quad . \quad . \quad . \quad (6)$$

The omission by Dr. Grzedzielski of the stress in the reinforcement from the final distribution was pointed out by us in our previous note as the correction to his equation (17) in Ref. 1. But it seems we were not sufficiently explicit, for in his reply Dr. Gzedzielski dismissed it as an "incorrect algebraic manipulation."

An independent verification of the consistency of the above arguments is obtained by checking the compatibility conditions in the modified structure. The condition to be satisfied is

$$\mathbf{b}_1{}'\mathbf{f}_m\mathbf{S}_m=\mathbf{0} \quad . \quad . \quad . \quad (7)$$

Substituting for $\mathbf{f}_m$ from equation (1) and for $\mathbf{S}_m$ from equation (6) we find without difficulty that the left hand side of equation (7) reduces to a zero matrix. Hence we may conclude that the determination of the stress distribution in the modified structure as presented is entirely consistent with the independently derived flexibility matrix of equation (1), and represents a correct version of the perturbation analysis.

But we must emphasise that the perturbation analysis, even in its correct form, gives no evidence to contradict the result obtained by the initial strain method. If they are applied to the same problem, they must yield the same final result. However, in this respect, the perturbation method is more limited in its applicability; at least in cases where the modified elements are coupled to the other structure.

As an example of this, it is not possible, in the idealised structure shown in Figs. 1 and 2, to modify a simple flange element, say 4,5. This is because the idealisation and the form of the flexibility matrix always suppose equal flange loads in the two flanges 4,5 and 5,6 at the point, 5, where they meet. The perturbation loads must retain this feature and hence the simplest perturbation system we can apply consists of a pair of equal and opposite shears along the two flange elements. Whether we like it or not, it is both elements which are being reinforced. If we only calculate the reinforcement flexibility for one element 4,5, it means that that for the other element is zero, *i.e.* it is rigid. We are, in effect, applying a combined reinforcement over the two elements, one part of the combination being rigid, while the other is flexible. This does not mean that the supposedly unreinforced element becomes rigid but rather that the effect of the peculiar reinforcement is averaged over the two elements by the way in which the compatibility conditions between them and the reinforcement are formed (equality of strains $\mathbf{v}_h$).

The difficulty here may be traced to the fact that the whole grid of effective (or lumped) flanges has been considered as a single structural element. It is moreover a redundant system, the redundancies being required to ensure compatibility of cross-strains at the flange intersections. Effectively then, we have to deal with a statically indeterminate basic system, but without explicit knowledge or discussion of the secondary redundancies. Though this may be of no consequence for the analysis of fairly uniform structures with no cut-outs, it brings difficulties when we wish to discuss cut-outs and reinforcement limited to one or only a few grid fields. To overcome these difficulties requires properly a fuller breakdown of the structure into simpler elements with the explicit introduction of additional redundancies to ensure compatibility of cross-strains. Such an approach was, in fact, adopted in the idealisation

used by Falkenheiner in Ref. 11. The coupling terms now disappear from the flexibility matrix itself since their effect is brought in by the solution of these secondary redundancies. The general theory of these and, in particular, their treatment in the analysis of modifications is given in Ref. 7.

An important computational advantage of eliminating coupling by introducing secondary redundancies is that the flexibility matrix **f** usually reduces to a "banded" structure which can be dealt with more efficiently in the computer. Some coupling terms remain, of course, but these are due to a statically determinate "series" combination of elements. No secondary redundancies are involved and the elements can therefore always be separated again for dealing with cut-outs and modifications. Thus the "three-banded" flexibility matrices of the "chain" elements used in Ref. 7 can always be broken down again into a diagonal assembly of simple element flexibilities for the treatment of local modifications. This avoids the difficulty mentioned two paragraphs back in dealing with a single element.

We hasten to add that it is not always necessary or even possible completely to eliminate the coupling. The case of a pre-calculated flexibility matrix for a sub-assembly such as a fuselage diaphragm is an obvious instance. But we must observe that there is rather more to the question than merely a few off-diagonal terms in the flexibility matrix, as appears from Dr. Grzedzielski's remarks. Again we should like to draw attention to the fact that his calculation of the flexibility matrix in his published work is essentially presented as the approximate evaluation of strain energy coefficients in terms of a finite number of stresses in a continuous sheet. It is not there clear just how actual flanges and stringers are to be dealt with (in fact no actual values of flexibilities are given) but the narrowness of view engendered by this approach certainly hinders a wider appreciation of the significance behind these off-diagonal terms.

5. CONCLUSIONS

We have shown that the analysis of structural cut-outs and modifications, using the initial strain concept, is correct in that the stress distributions it predicts satisfy the compatibility conditions in the cut or modified structure, and that this applies regardless of any cross-coupling in the flexibility matrix.

The criticisms of Dr. Grzedzielski are shown to be unfounded and his modification analyses seriously in error. His analysis of cut-outs, though algebraically without error, yields a stress distribution which is shown to be incompatible in the final structure. Considering that the possible coupling in the flexibility matrix is the main point of Dr. Grzedzielski's thesis, it is surprising what little justice he does to its real effect and significance.

The method of thermal stress analysis proposed by Dr. Grzedzielski, which he claims to be more accurate than ours, is also shown to be inconsistent with the predictions of the Theory of Elasticity as far as the effect of Poisson's ratio is concerned.

We must finally observe that, at no point in his argument, has Dr. Grzedzielski put forward positive evidence to support his claims. He has merely presented an (incorrect) argument leading to results different from ours and thereafter appealed irrelevantly to principle.

APPENDIX A

ANALYSIS OF A FLAT, ISOTROPIC SHEET BY THE MATRIX FORCE METHOD

Figure 1(*a*) shows a flat, rectangular sheet of isotropic elastic material. Our aim is to analyse the stresses in this sheet by the matrix force method. To this end, we idealise it into the singly redundant grid system of six direct stress-carrying flanges and four shear-carrying fields of Fig. 1(*b*). The idealisation is admittedly crude, but the essential conclusions are unaffected by the choice of a finer grid.

If the sheet is of thickness t, the effective flange areas may be taken, on the simplest approximation, as $bt/2$ and $at/2$ for the outer flanges in the x- and y- directions respectively and as bt and at for the inner flanges. With the numbering system for flange loads and shear flows indicated in Fig. 2, the flexibility matrix $\mathbf{f}_l$ of the idealised flanges, as given by Dr. Grzedzielski is

$$\mathbf{f}_l=\frac{1}{6Et}\begin{bmatrix}\frac{a}{b}\mathbf{A} & -6\nu\mathbf{B}\\ -6\nu\mathbf{B} & \frac{b}{a}\mathbf{A}\end{bmatrix} \quad . \quad . \quad (A1)$$

where the square, symmetric matrices **A, B** are

$$\mathbf{A}=\begin{bmatrix}4&2&.&.&.&.&.&.&.&.&.&.\\2&8&2&.&.&.&.&.&.&.&.&.\\.&2&4&.&.&.&.&.&.&.&.&.\\.&.&.&2&1&.&.&.&.&.&.&.\\.&.&.&1&4&1&.&.&.&.&.&.\\.&.&.&.&1&2&.&.&.&.&.&.\\.&.&.&.&.&.&4&2&.&.&.&.\\.&.&.&.&.&.&2&8&2&.&.&.\\.&.&.&.&.&.&.&2&4&.&.&.\end{bmatrix} \quad \mathbf{B}=\begin{bmatrix}1&.&.&.&.&.&.&.&.&.&.&.\\.&.&.&1&.&.&.&.&.&.&.&.\\.&.&.&.&.&.&1&.&.&.&.&.\\.&1&.&.&.&.&.&.&.&.&.&.\\.&.&.&1&.&1&.&.&.&.&.&.\\.&.&.&.&.&.&1&.&1&.&.&.\\.&.&1&.&.&.&.&.&.&.&.&.\\.&.&.&1&.&.&.&1&.&.&.&.\\.&.&.&.&.&.&1&.&.&.&1&.\end{bmatrix} \quad (A2)$$

in which elements indicated by a dot are zero.

The flexibility $\mathbf{f}_q$ of the shear panels is

$$\mathbf{f}_q=\frac{ab}{Gt}\mathbf{I}_4 \quad . \quad . \quad (A3)$$

where $\mathbf{I}_4$ is the unit matrix of order 4.

As the single redundancy, we choose an **X** system similar to those of Refs. 4, 5, 6, but scaled to give simpler

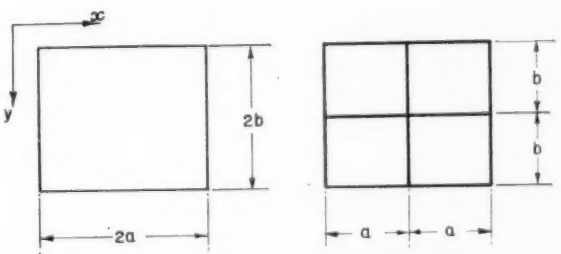

FIGURE 1. Idealised rectangular plate.

FIGURE 2. Numbering of flange loads and shear flows in idealised plate.

quantities for the present investigation. The matrices $\mathbf{b}_{1i}$, $\mathbf{b}_{1q}$ of flange loads and shear flows due to unit redundancy are

$$\left.\begin{aligned}\mathbf{b}_{1i}&=\{0\ -a\ 0\ 0\ 2a\ 0\ 0\ -a\ 0\ 0\ -b\ 0\ 0\ 2b\ 0\ 0\ -b\ 0\}\\ \mathbf{b}_{1q}&=\{1\ -1\ -1\ 1\}\end{aligned}\right\}\quad (A4)$$

The (1×1) matrix

$$\mathbf{D}=\mathbf{b}_1{}^t\mathbf{f}\mathbf{b}_1=\mathbf{b}_{1i}{}^t\mathbf{f}_i\mathbf{b}_{1i}+\mathbf{b}_{1q}{}^t\mathbf{f}_q\mathbf{b}_{1q}\quad .\quad (A5)$$

which is required in the analysis, is now easily evaluated as

$$\mathbf{D}=\frac{8}{3Eabt}\{2a^4-3\nu a^2b^2+2b^4\}+\frac{4ab}{Gt}\quad .\quad (A6)$$

But if we substitute for G from the relation

$$E=2G(1+\nu)\quad .\quad .\quad (A7)$$

for an isotropic material, we find that the terms in ν cancel out and there remains only

$$\mathbf{D}=\frac{8}{3Eabt}\{2a^4+3a^2b^2+2b^4\}\quad .\quad .\quad (A8)$$

We may note here that the imposition of a kinematic constraint, say by building in one of the edges of the plate, would require additional redundancies. In general, Poisson's ratio would then no longer cancel out of the **D** matrix.

Suppose now that the grid system is loaded by edge and body forces which give rise to the statically equivalent system of flange loads and shear flows described by the (22×1) column matrix $\mathbf{S}_0$. The final stress system, satisfying also compatibility conditions, is then

$$\mathbf{S}=\mathbf{S}_0-\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_1{}^t\mathbf{f}\mathbf{S}_0\quad .\quad .\quad (A9)$$

Since **D** is independent of ν, the final stress distribution will also be independent of ν provided it cancels out of the term $\mathbf{b}_1{}^t\mathbf{f}\mathbf{S}_0$. Distinguishing with a suffix ν the part of this term dependent on Poisson's ratio, we find

$$(\mathbf{b}_1{}^t\mathbf{f}\mathbf{S}_0)_\nu=\frac{\nu}{Et}\{b(S_4-2S_5+S_6)+a(S_{13}-2S_{14}+S_{15})\}+$$

$$+\frac{\nu ab}{Et}\{S_{19}-S_{20}-S_{21}+S_{22}\}\quad .\quad (A10)$$

But, if we introduce ω_{ij} as the body force per unit length along the flange element ij, we can obtain by combination of the equilibrium conditions of two neighbouring flange elements:

$$S_4-2S_5+S_6=-a(S_{19}-S_{20}-S_{21}+S_{22})-a(\omega_{5,6}-\omega_{4,5})\quad .\quad .\quad .\quad (A11)$$

with a similar relation for flange 13, 14, 15.

Substituting these in (*A*10) gives therefore

$$(\mathbf{b}_1{}^t\mathbf{f}\mathbf{S}_0)_\nu=-\frac{\nu ab}{Et}\{\omega_{5,6}-\omega_{4,5}+\omega_{14,15}-\omega_{13,14}\}\quad .\quad (A12)$$

If therefore the body forces ω_{ij} are constant along each flange, equation (*A*12) reduces to zero and Poisson's ratio disappears entirely from the results of the analysis. The calculated stress distribution thus retains the independence of the material elastic properties demonstrated by exact analysis.

Since in our analysis of thermal stresses, **H** does not involve Poisson's ratio, it is obvious that the final distribution of thermal stress,

$$\mathbf{S}=-\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_1{}^t\mathbf{H}\quad .\quad .\quad .\quad (A13)$$

is, by this method, also independent of Poisson's ratio.

But, as calculated by Dr. Grzedzielski, **H** does involve ν. It remains to see whether it disappears in the term $\mathbf{b}_1{}^t\mathbf{H}$.

The matrix $\mathbf{b}_{0\theta}$ for the present problem, with allowance for the differing effective flange areas, is

$$\mathbf{b}_{0\theta}=\frac{\alpha tE}{2(1-\nu)}\left[\begin{array}{c|c|c}
b & \begin{matrix}1&&\\&1&\\&&1\end{matrix} & \begin{matrix}&&\\&&\\&&\end{matrix}\\
\hline
 & \begin{matrix}2&&\\&2&\\&&2\end{matrix} & \begin{matrix}&&\\&&\\&&\end{matrix}\\
\hline
 & \begin{matrix}&&\\&&\\&&\end{matrix} & \begin{matrix}1&&\\&1&\\&&1\end{matrix}\\
\hline
a & \begin{matrix}1&&\\&1&\\&&1\end{matrix} & \begin{matrix}&&\\&&\\&&\end{matrix}\\
\hline
 & \begin{matrix}2&&\\&2&\\&&2\end{matrix} & \begin{matrix}&&\\&&\\&&\end{matrix}\\
\hline
 & \begin{matrix}1&&\\&1&\\&&1\end{matrix} & \begin{matrix}&&\\&&\\&&\end{matrix}
\end{array}\right]\quad .\quad (A14)$$

where the numbering of temperatures, $\theta_1\ \ldots\ \theta_9$, is the same as for the first nine flange loads.

Using (*A*14) we find, by Dr. Grzedzielski's analysis,

$$\mathbf{b}_1{}^t\mathbf{H}=\mathbf{b}_1{}^t\mathbf{f}\mathbf{b}_{0\theta}\boldsymbol{\theta}=$$

$$\frac{\alpha(a^2+b^2)}{6(1-\nu)}\{-1\ \mu\ -1\ \eta\ -(\mu+\eta)\ \eta\ -1\ \mu\ -1\}\boldsymbol{\theta}\quad (A15)$$

where

$$\mu=\frac{2b^2-4a^2+6a^2\nu}{a^2+b^2}\quad ,\quad \eta=\frac{2a^2-4b^2+6b^2\nu}{a^2+b^2}\quad .\quad (A16)$$

We conclude, therefore, that for a general temperature distribution, the stresses calculated this way will show a dependence on Poisson's ratio. This conclusion is not affected by a more refined example. We may, for example, set up the difference equations, in the redundancies X for a regular grid, according to the two methods. It is of interest here that the values of X are effectively point values of an Airey stress function for the plate. The difference equation derived in this way by Dr. Grzedzielski's argument still contains terms involving ν.

APPENDIX B

COMPATIBILITY CONDITIONS IN THE STRUCTURE WITH CUT-OUTS

The analysis by the initial strain approach[4], [5], [6] yields the following expression for the stress distribution in the structure with cut-outs.

$$\mathbf{S}_c=\mathbf{b}_c\mathbf{R}\quad .\quad .\quad .\quad (B1)$$

where $$\mathbf{b}_c=\mathbf{b}-\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t]^{-1}\mathbf{b}_h\quad .\quad (B2)$$

The matrix **b** gives the stress distribution in the original,

uncut structure and hence satisfies the compatibility condition

$$\mathbf{b}_1{}^t\mathbf{f}\mathbf{b}=\mathbf{0} \quad . \quad . \quad . \quad (B3)$$

The matrix $\mathbf{b}_e$ of equations (*B*1) and (*B*2) includes also the h zero stresses in the cut-out elements. Though useful as a numerical check in practical calculations, these zeros are not required here. The stress distribution in the remaining (g) structure is

$$\mathbf{b}_{cg}=\mathbf{b}_g-\mathbf{b}_{1g}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t]^{-1}\mathbf{b}_h \quad . \quad (B4)$$

We write now the flexibility matrix of the original structure as

$$\mathbf{f}=\begin{bmatrix}\mathbf{f}_g & \mathbf{f}_{gh}\\ \mathbf{f}_{hg} & \mathbf{f}_h\end{bmatrix} \quad . \quad . \quad . \quad . \quad (B5)$$

The structure remaining after the elimination of the h-elements is defined by the flexibility matrix $\mathbf{f}_g$. The degree of redundancy of this structure is clearly less than that of the original structure. In fact, if n is the number of original redundancies, the number in the structure with cut-outs must be $n-h$, where h is the number of stress resultants which have to be nullified in order to obtain the cut-outs. It is clear that the original n redundancies $\mathbf{X}$ are not directly suitable for an investigation of the cut structure since they give rise, in general, to stresses in the cut-out elements. But they can be used to define a suitable, smaller set of $(n-h)$ redundancies $\mathbf{Y}$ by the transformation

$$\mathbf{X}=\mathbf{B}\mathbf{Y} \quad . \quad . \quad . \quad (B6)$$

where $\mathbf{B}$ is an $n\times(n-h)$ transformation matrix.

Since the new redundancies $\mathbf{Y}$ must not give rise to stresses in the cut-out elements, $\mathbf{B}$ must satisfy the condition

$$\mathbf{b}_{1h}\mathbf{B}=\mathbf{0} \quad . \quad . \quad . \quad (B7)$$

Provided equation (*B*7) is satisfied, the choice of $\mathbf{B}$ is arbitrary and does not interest us further.

The set of self-equilibrating stress systems for the structure with cut-outs, due to unit values of the redundancies $\mathbf{Y}$, is clearly

$$\mathbf{b}_{1g}\mathbf{B}$$

and the compatibility condition for this structure may now be written as

$$\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_g\mathbf{b}_{cg}=\mathbf{0} \quad . \quad . \quad . \quad (B8)$$

Equation (*B*8) is exactly analogous to equation (*B*3) for the original structure. Substituting for $\mathbf{b}_{cg}$ from (*B*4) we obtain

$$\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_g\mathbf{b}_g-\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_g\mathbf{b}_{1g}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t]^{-1}\mathbf{b}_h=\mathbf{0} \quad . \quad . \quad . \quad . \quad . \quad (B9)$$

To show that this condition is satisfied we make use of equation (*B*5) and the corresponding partitioning of $\mathbf{b}$, $\mathbf{b}_1$ to write equation (*B*3) as

$$\mathbf{b}_1{}^t\mathbf{f}\mathbf{b}=\mathbf{b}_{1g}{}^t\mathbf{f}_g\mathbf{b}_g+\mathbf{b}_{1g}{}^t\mathbf{f}_{gh}\mathbf{b}_h+\mathbf{b}_{1h}{}^t\mathbf{f}_{hg}\mathbf{b}_g+\mathbf{b}_{1h}{}^t\mathbf{f}_h\mathbf{b}_h=\mathbf{0} \quad . \quad . \quad . \quad (B10)$$

We also require the corresponding form for $\mathbf{D}$.

$$\mathbf{D}=\mathbf{b}_1{}^t\mathbf{f}\mathbf{b}_1=\mathbf{b}_{1g}{}^t\mathbf{f}_g\mathbf{b}_{1g}+\mathbf{b}_{1g}{}^t\mathbf{f}_{gh}\mathbf{b}_{1h}+\mathbf{b}_{1h}{}^t\mathbf{f}_{hg}\mathbf{b}_{1g}+\mathbf{b}_{1h}{}^t\mathbf{f}_h\mathbf{b}_{1h} \quad . \quad . \quad (B11)$$

Substituting for $\mathbf{b}_{1g}{}^t\mathbf{f}_g\mathbf{b}_g$ from (*B*10) and for $\mathbf{b}_{1g}{}^t\mathbf{f}_g\mathbf{b}_{1g}$ from (*B*11) we find now for the left hand side of (*B*8) or (*B*9)

$$\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_g\mathbf{b}_{cg}=-\mathbf{B}^t[\mathbf{b}_{1g}{}^t\mathbf{f}_{gh}\mathbf{b}_h+\mathbf{b}_{1h}{}^t\mathbf{f}_{hg}\mathbf{b}_g+\mathbf{b}_{1h}{}^t\mathbf{f}_h\mathbf{b}_h]-$$
$$-\mathbf{B}^t\mathbf{b}_{1h}{}^t[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t]^{-1}\mathbf{b}_h+$$
$$\mathbf{B}^t[\mathbf{b}_{1g}{}^t\mathbf{f}_{gh}\mathbf{b}_{1h}+\mathbf{b}_{1h}{}^t\mathbf{f}_{hg}\mathbf{b}_{1g}+\mathbf{b}_{1h}{}^t\mathbf{f}_h\mathbf{b}_{1h}]\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t]^{-1}\mathbf{b}_h \quad . \quad . \quad . \quad . \quad (B12)$$

Most of the terms in (*B*12) vanish immediately because of the condition (*B*7). There remains

$$\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_g\mathbf{b}_{cg}=$$
$$-\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_{gh}\mathbf{b}_h+\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_{gh}\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t]^{-1}\mathbf{b}_h$$
$$=\mathbf{0}$$

Thus, the stress distribution of equation (*B*2) or (*B*4) does satisfy exactly the compatibility conditions in the structure with cut-outs.

The stress distribution derived by Dr. Grzedzielski in Ref. 1, (equation 11) for the cut-out case is

$$\mathbf{b}_{cg}=\mathbf{b}_g-\mathbf{b}_{1g}\mathbf{D}^{-1}\mathbf{b}_1{}^t\mathbf{f}_h{}^*[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_1{}^t\mathbf{f}_h{}^*]^{-1}\mathbf{b}_h \quad (B13)$$

where

$$\mathbf{f}_h{}^*=\{\mathbf{f}_{gh}\ \mathbf{f}_h\} \quad . \quad . \quad . \quad (B14)$$

Using this result in equation (*B*8) and following the same argument as above, we find that the left hand side of equation (*B*8), after using (*B*7), reduces to

$$\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_g\mathbf{b}_{cg}=-\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_{gh}\mathbf{b}_h-\mathbf{B}^t\mathbf{b}_1{}^t\mathbf{f}_h{}^*[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_1{}^t\mathbf{f}_h{}^*]^{-1}\mathbf{b}_h+$$
$$+\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_{gh}[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_1{}^t\mathbf{f}_h{}^*][\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_1{}^t\mathbf{f}_h{}^*]^{-1}\mathbf{b}_h$$
$$=-\mathbf{B}^t\mathbf{b}_1{}^t\mathbf{f}_h{}^*[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_1{}^t\mathbf{f}_h{}^*]^{-1}\mathbf{b}_h$$
$$=-\mathbf{B}^t\mathbf{b}_{1g}{}^t\mathbf{f}_{gh}\mathbf{g} \quad . \quad . \quad . \quad . \quad (B15)$$

The last reduction is achieved by substituting for the first $\mathbf{f}_h{}^*$ from (*B*14) and making use again of (*B*7). Also $\mathbf{g}$ is the matrix of perturbation forces $\mathbf{G}$ (equation (10), Ref. 1) for unit values of the external loads $\mathbf{R}$.

$$i.e. \quad \mathbf{G}=\mathbf{g}\mathbf{R}. \quad . \quad . \quad (B16)$$

It is clear that (*B*15) is not zero since $\mathbf{f}_{gh}\mathbf{g}$ is the matrix of strains in the remaining (g) structure due to the perturbation forces. There is no reason why they should be compatible. Indeed it is a main point of Dr. Grzedzielski's argument that they are not.

APPENDIX C

COMPATIBILITY CONDITIONS IN THE MODIFIED STRUCTURE

We have to show here that the stress distribution

$$\mathbf{S}_m=\mathbf{b}_m\mathbf{R} \quad . \quad . \quad . \quad . \quad (C1)$$

where

$$\mathbf{b}_m=\mathbf{b}-\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t+\Delta\mathbf{f}_h{}^{-1}]^{-1}\mathbf{b}_h \quad . \quad (C2)$$

calculated by the initial strain approach to the modification problem, is correct in the structure with flexibility matrix $\mathbf{f}_m$, where

$$\mathbf{f}_m=\mathbf{f}+\begin{bmatrix}\mathbf{0} & \mathbf{0}\\ \mathbf{0} & \Delta\mathbf{f}_h\end{bmatrix} \quad . \quad . \quad . \quad (C3)$$

The proof is straightforward since we may here use directly the original redundancies of the unmodified structure in order to express the compatibility conditions. These are

$$\mathbf{b}_1{}^t\mathbf{f}_m\mathbf{b}_m=\mathbf{0} \quad . \quad . \quad . \quad (C4)$$

or, substituting for $\mathbf{f}$ from (*C*3),

$$\mathbf{b}_1{}^t\mathbf{f}\mathbf{b}_m+\mathbf{b}_{1h}{}^t\Delta\mathbf{f}_h\mathbf{b}_{mh}=\mathbf{0} \quad . \quad . \quad (C5)$$

With $\mathbf{b}_m$ and its submatrix $\mathbf{b}_{mh}$ given by equation (*C*2) we find for the left hand side of equation (*C*4)

$$\mathbf{b}_1{}^t\mathbf{f}_m\mathbf{b}_m=\mathbf{b}_1{}^t\mathbf{f}\mathbf{b}-\mathbf{b}_1{}^t\mathbf{f}\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t+\Delta\mathbf{f}_h{}^{-1}]^{-1}\mathbf{b}_h+$$

$$+\mathbf{b}_{1h}{}^t\Delta\mathbf{f}_h\mathbf{b}_h-\mathbf{b}_{1h}{}^t\Delta\mathbf{f}_h\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t+\Delta\mathbf{f}_h{}^{-1}]^{-1}\mathbf{b}_h.$$

The first term vanishes since it is the compatibility condition satisfied by $\mathbf{b}$ in the unmodified structure (equation (*B*3)). Noting also that

$$\mathbf{b}_1{}^t\mathbf{f}\mathbf{b}_1=\mathbf{D}$$

we can now combine the second and fourth terms and finally obtain:

$$\mathbf{b}_1{}^t\mathbf{f}_m\mathbf{b}_m=-\mathbf{b}_{1h}{}^t\Delta\mathbf{f}_h[\Delta\mathbf{f}_h{}^{-1}+\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t]\times$$

$$\times[\mathbf{b}_{1h}\mathbf{D}^{-1}\mathbf{b}_{1h}{}^t+\Delta\mathbf{f}_h{}^{-1}]^{-1}\mathbf{b}_h+\mathbf{b}_{1h}{}^t\Delta\mathbf{f}_h\mathbf{b}_h$$

$$=-\mathbf{b}_{1h}{}^t\Delta\mathbf{f}_h\mathbf{b}_h+\mathbf{b}_{1h}{}^t\Delta\mathbf{f}_h\mathbf{b}_h$$

$$=\mathbf{0}$$

Hence the compatibility conditions (*C*4) are exactly satisfied and we conclude that equation (*C*2) gives the correct stress distribution in the structure with modifications defined by equation (*C*3).

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Closing Comment on the Initial Strain Concept

ALEX. L. M. GRZEDZIELSKI, Dr.Eng., P.Eng.

(*Note*: In order to save space, the reference numbers in this note are those of the previous note (on p. 129) and the formulae are numbered consecutively.)

WITH only a thousand words allowed for an answer to Messrs. Argyris' and Kelsey's lengthy contribution, I can touch barely on the most important aspects of the controversy. According to Professor Argyris and Mr. Kelsey there are two versions of the Castigliano Theorems: an original and narrow version which assumes all strain elastic, and to which I am supposed to be dedicated; and the modern version, based on the conditions of compatibility of total strains, in which the initial (or thermal) strains are treated exactly in the same manner as the elastic ones. I have nothing against generalisation of the old masters' theorems whenever this is appropriate, but I do object strenuously to any concept implying that initial strain can be imposed at will on structural elements. In my understanding this amounts to writing the strain-stress law in the form

$$\eta_{xx}=[\sigma_{xx}-\nu(\sigma_{yy}+\sigma_{zz})]/E+\alpha\theta+\text{whatever required}$$

which I cannot consider as a serious proposition in any purely elastic analysis.

As concerns the thermal stress analysis I have adopted the approach of Refs. 4 and 5 where the solution is presented in terms of fictitious body and surface forces depending on temperature. This concept is, in my opinion, particularly suited for the Matrix Force Method. It fits into the matrix formulation of this method and, since the temperature effect is presented in terms of forces, a use can be made of all general theorems and methods of structural analysis. I hope to have shown in Ref. 7 to my previous note (p. 129), and in greater detail in this note, that the fictitious loads enter the computation by medium of a special $\mathbf{b}_{0\theta}$-matrix, relating temperature to the statically determinate part of stress distribution.

The Argyris-Kelsey formulation actually is not very much off. It is simple and yields correct results in all truss and frame structures where the lateral contraction of the material ν does not appear and where the sweep effect does not exist. However, their approach is not adequate, in my opinion, in a multi-spar-rib swept structure where an

intricate coupling exists between various stress components, as indicated by the following strain-stress law of an oblique panel and which somehow has to be accounted for by the general flexibility matrix of the wing:

$$\begin{vmatrix} \varepsilon_{xx} \\ \varepsilon_{yy} \\ \varepsilon_{xy} \end{vmatrix} = \frac{1}{E \sin^2 \alpha} \begin{vmatrix} 1 & \cos^2 \alpha - \nu \sin^2 \alpha & 2 \cos \alpha \\ \cos^2 \alpha - \nu \sin^2 \alpha & 1 & 2 \cos \alpha \\ 2 \cos \alpha & 2 \cos \alpha & 2[(1+\nu) \sin^2 \alpha + 2 \cos^2 \alpha] \end{vmatrix} \times \begin{vmatrix} \sigma_{xx} \\ \sigma_{yy} \\ \sigma_{xy} \end{vmatrix} \quad (12)$$

Here the off diagonal terms express the Poisson's ratio coupling and, those depending on α only, the sweep coupling. It is plain that one cannot be discussed without the other as has been done erroneously by Professor Argyris and Mr. Kelsey in their note above. I should point out that in the example of my last note I was not considering a flat isotropic sheet, as reported by Argyris and Kelsey, but a portion of a wing plate where the Poisson's ratio does *not disappear*.

The inferiority of the initial strain concept can be demonstrated directly in the case of the structural modification method. However, since all my work in this respect is declared invalid by Argyris and Kelsey and I have not enough space left for comments, I should like to use in my discussion the few equations on which we all three can agree. These are the equations (1) to (4) of Professor Argyris' and Mr. Kelsey's last instalment.

The argument can be stated as follows: when equation (4) is multiplied by $\mathbf{f}_h(\mathbf{f}_h+\mathbf{f}_{ah})^{-1}$ it yields, in view of (1) and in agreement with the definition of the $\mathbf{f}_h^*$ -matrices, the former equation (9) in a slightly modified form:

$$\mathbf{f}_h\mathbf{G}=(\mathbf{f}_{hm}^{*t}-\mathbf{f}_h^{*t})\,[\mathbf{bR}-\mathbf{b}_1\mathbf{D}^{-1}\mathbf{b}_1{}^t\mathbf{f}_h^*\mathbf{G}]. \quad (13)$$

For further proof this equation is to be compared with equation (66) of Ref. 1, rewritten above as equation (10). In order to be more specific assume that the structural modification consists of the addition of some flange (cap) material at a certain spar-rib intersection. The $\mathbf{f}_h^*$ -submatrix required is then, by (12),

$$\mathbf{f}_h^{*t}=\begin{vmatrix} \ldots(E\nu\alpha)\ldots.(E)\ldots.(E\alpha)\ldots.(E\alpha)\ldots.(E\alpha)\ldots.(E\alpha)\ldots \\ \ldots(E)\ldots.(E\nu\alpha)\ldots(E\alpha)\ldots.(E\alpha)\ldots.(E\alpha)\ldots.(E\alpha)\ldots \end{vmatrix} \quad (14)$$

where (E) represents the two flange terms on the main diagonal, $(E\nu\alpha)$ the Poisson's ratio couplings, $(E\alpha)$ the sweep couplings of the flange stresses with the four adjacent panel shears. The $\mathbf{f}_{ah}$ -matrix of the present case is a 2×2 diagonal. The **G** loads may be visualised as a uniform surface shear, applied along the flanges, so as to generate direct stresses of a given magnitude at the spar-rib intersection.

It becomes quite obvious now that equations (13) and (10) become identical if the $(E\nu\alpha)$ and $(E\alpha)$ terms vanish in (14), because then the transformation (11), $\mathbf{H}=\mathbf{f}_h\mathbf{G}$, will hold. This is an important step forward, for it demonstrates without any doubt that the initial strain concept is, indeed, a special case of the **G**-load approach.

Unfortunately, our accord ceases with equation (4), and I cannot agree with the further statement of Messrs. Argyris and Kelsey that "The stresses in the final structure must include those in the reinforcement and hence a term $-\mathbf{G}$ has to be added to the second equation (17)." According to the derivation of (4),

(*a*) the original structure is loaded by **R** and **G**,
(*b*) the reinforcement is loaded by $-\mathbf{G}$ only,
(*c*) the **G**-loads are such that the strains are matched.

Thus I ask the question: From where should the term **G** be subtracted and why? Is it not that stresses in the reinforcement and in the original structure should be the same?

I hope sincerely that Professor Argyris and Mr. Kelsey will be inclined to reconsider their last reservation. If they were to do so, I am sure that all aspects of this difficult problem would be resolved and that the issue could be considered as settled. I wish that a similar ground for common understanding could also be found for the thermal stress analysis. May I suggest that the formulation of the Castigliano Theorem due to Hemp (A.R.C. Struc. 1683) may bring the clarification.

Concluding Comments on the Initial Strain Concept

J. H. ARGYRIS and S. KELSEY

IN PRESENTING our final comments we should like first to recall the origin and basis of this rather prolonged argument. The first note by Dr. Grzedzielski objected to the concept of initial strain and its use in two instances —thermal stress analysis and the treatment of structural modifications and cut-outs. In particular, he showed (after some clarification of notation) that the methods he proposed could, in some circumstances, yield *different* algebraic results from ours. But we must note that the inadequacy or inferiority of the initial strain method inferred as a result by Dr. Grzedzielski represents his opinion and preference; they are not substantiated by the presentation of any definite evidence direct or indirect. In fact his modification analysis contained (and still contains) serious errors and his cut-out result is demonstrably incorrect.

As far as thermal stresses are concerned, the issue reduces to the question of how the quantities **H** are to be calculated. The question of nomenclature is here not important, for however they are calculated, (whether directly or as $\mathbf{fb}_0\boldsymbol{\theta}_0$), these quantities must represent the generalised thermal strains of the statically determinate basic system. By definition, they are the initial strains of the particular problem.

According to Dr. Grzedzielski, the direct calculation of **H**, by application of the unit load theorem to the detailed thermal strain distribution, is not adequate for the cover of a swept, low aspect ratio wing and more generally where the flexibility matrix **f** displays a large number of coupling terms. But it is not made clear why and how this should be so; it is only shown that the method of fictitious thermal loads gives a different result.

As far as we are aware, the only positive evidence on the relative accuracy of the two methods is the example which we gave in our second note (Appendix A). This example is confined to the effect of Poisson's ratio and, although in practical problems the difference is likely to be small, this simple example does indicate a more consistent analysis by the direct calculation of **H**. At any rate there is no suggestion of inadequacy. Dr. Grzedzielski now insists it is not correct to separate the effects of sweep and Poisson's ratio. While this is no doubt true for a swept wing, one wonders at what angles of sweep the supposed inadequacy and superiority are going to show themselves and precisely what evidence there is.

We should like to point out also that we have never challenged the necessity of unusual couplings due to swept structures or of thermal strains in swept panels. Our point is that Dr. Grzedzielski's is not the only approach to the problem and that the form of the flexibility matrix has a connection with the generalised thermal strains only when these are calculated by the method of fictitious thermal loads. Differences in the results by the two methods arise because we are dealing with an idealised structure. We repeat that the application of a generalised version of the Castigliano Theorem gives results in agreement with ours, because such a theorem is effectively only a restatement of the compatibility conditions on the total strains. In practical calculations, the framing of these conditions as a minimum energy statement is unnecessary, serving only to complicate the problem and obscuring the unity and simplicity of the fundamental compatibility conditions given directly by the Virtual Forces Principle. But in any case the method of fictitious thermal loads does not follow directly from such a generalised theorem. It requires further manipulation; first to introduce a stiffness in order to convert the initial (thermal) strains into initial stresses and second a flexibility to reconvert these to generalised initial strains. In the idealised structure the above stiffness and flexibility may not be perfectly reciprocal, as is the case in the example of our previous note, and it is in this that the basic cause of difference appears to lie.

In the case of cut-outs and modifications we did not declare the whole of Dr. Grzedzielski's work to be invalid. We proved his analysis to be wrong in certain important respects and corrected his comparative interpretation of the results. We also devoted considerable space to proving directly the validity of the initial strain approach. The raising of this from the level of incorrectness to that of inferiority only to the perturbation analysis presumably constitutes an acknowledgment of its validity. It is gratifying to see some measure of agreement in the perturbation method itself, but we must observe that the equations now agreed are not those originally given by Dr. Grzedzielski, and that the "slight modifications" (apparently accepted by him) are of the same order as those which presumably caused Dr. Grzedzielski to write his original note. It is, too, somewhat unusual to describe one method as a special case of another for the sole reason that they give identical results in a particular set of circumstances. This identity has, moreover, been clear from the start and can hardly be called an "important step forward."

Unfortunately, we must insist on the whole of our version of the perturbation analysis. In the simple case of a reinforced shear panel—Dr. Grzedzielski's original example—the shear flow in the final panel, which is what we wish to calculate, is equal to that in the original panel (due to loads **R** and **G**) plus that in the reinforcement ($-\mathbf{G}$). This is why the term **G** must be subtracted; our previous note shows quite clearly where. It should in any case be obvious that the changes in the stress distribution (note the generalised meaning of stress here) should be expressible in the form

$$\mathbf{S}_m - \mathbf{S} = \mathbf{b}_1 \mathbf{X}_m$$

where $\mathbf{X}_m$ is the column of changes to the redundancies. Thus a single closed expression is possible for $\mathbf{S}_m$, which is not the case in Dr. Grzedzielski's argument.

In view of the error still remaining in Dr. Grzedzielski's perturbation analysis it is not difficult to imagine significant differences in the results of practical problems by the two approaches. It is however a little more difficult to visualise a direct demonstration of the superiority of his approach.

Finally, we emphasise the generalised nature of the initial strains **H**. With this qualification and with the completion of Dr. Grzedzielski's strain-stress law (at the end of his first paragraph) to read "whatever required to simulate the additional flexibility of the cut-out or modification" we find the, to him unacceptable, interpretation of the initial strain approach correct.

While R.Ae.S. Branches are to be found in all the four countries of the U.K., BELFAST was the first that survives to be formed outside England. (If this seems a negative statement, let us positively state that it was the first, and is the only, Branch in Ireland).

THE Aircraft Industry, in the form of Short and Harlands, came to Northern Ireland in 1936 and by 1938 had expanded sufficiently for a number of enthusiasts to consider forming a Branch of the Society in Belfast.

At the first General Meeting on 23rd November 1938, a Committee was formed under the Chairmanship of Mr. J. H. Lower, Works Manager at Short and Harlands, to initiate Branch activities. The inaugural lecture, held in the Central Hall of Belfast College of Technology in January 1939, was by Mr. A. Gouge (now Sir Arthur), Chief Designer and General Manager of Short Brothers, and attracted an attendance of almost 400. When one considers that there were fewer than 10 parent body members in the area at that time, the first lecture attendance and a membership of around 180, were excellent achievements.

The main source of membership was Shorts but there was considerable interest from people whose occupations were not even remotely connected with engineering.

Two further lectures were given in the first session: war intervened and activities were suspended, with the exception of the library which remained open and was used considerably by members.

By 1944 it was decided that, as things had settled down to a certain degree, a resumption was possible and a General Meeting was called in October 1944, after which the Committee with Mr. W. Browning, Works Manager of Shorts, as Chairman, arranged a programme of lectures having local lecturers and attracting an average attendance of around 100, but reaching as high as 230 on one occasion.

Speakers in this first post-war series dealt progressively with aircraft design, production, and ended up with test flying. In 1946 the branch founded the Belfast Branch Prize as an annual award to the best student attending the post-higher national certificate course in aeronautics at the Belfast College of Technology and, also in 1946, Mr. Hambrook, who had been Honorary Secretary of the Branch since its inception, was elected Chairman.

With the transfer of Messrs. Short Brothers Seaplane Works from Rochester to Belfast, the large number of members of the Medway Branch of the Society (which was also formed in 1938) were transferred so that membership of that Branch fell sharply and at the Annual General Meeting on 22nd September 1947, it was decided to obtain the approval of the Council to amalgamate the Branch with the Belfast Branch; this was done in January 1948.

The main source from which members were drawn remained the aircraft factory, now Short Brothers and Harland, Limited, but with interested persons drawn from civil and service airfields not far distant from Belfast and Queen's University, the total was around 250.

Social activities were not forgotten. Visits were arranged and a highlight was an annual "smoker" where the usual attendance of 250 was limited only by the accommodation. These were succeeded by the annual dance which has been very successful for the past eight years.

When travel became easier after the war the scope of the lectures was greatly widened and among the notable cross-channel lecturers were Captain J. L. Pritchard on "Bird Flight and the Aeroplane" and Mr. P. G. Masefield on "Civil Aviation—Present and Future." Local lecturers included the late Professor G. T. R. Hill, who was attached to Shorts in a consultative capacity and who was one of the Branch's stalwarts. His lecture in the 1948-49 session "1,000 Miles Per Hour?" proved truly prophetic and of the greatest interest.

It has been the Branch's policy to change its Committee Chairmen frequently and in 1948 Rear Admiral M. S. Slattery (now Sir Matthew, Chairman of B.O.A.C.) became branch Chairman and held the post for some four years, after which he was elected President, which position he still holds.

Despite certain travel difficulties up to the beginning of the present session three Main Society lectures have been arranged. The first, on 7th November 1950, was given by Major P. L. Teed on "Aircraft Metallic Material Under Low Temperature Conditions." The second was held on 10th December 1953, when Mr. C. B. Ringham and Dr. A. E. Cutler lectured on "Training Devices in Aircraft" and will clearly be remembered by all who attended. The third, by Mr. G. Forrest, on "Problems Associated With Production and Use of Modern Light Alloys" was held on 5th April 1956.

Since May 1958, when the David Keir Building of Queen's University, Belfast was completed, the University authorities have been very kind in permitting the Branch to use one of the lecture theatres (seating about 250) for their lectures and, for special occasions, the Sir William Whitla Hall, which has a seating capacity of 1,400. Both of these are among the finest of their size in the U.K. and the Branch would like to take the opportunity to express its gratitude to the University Authorities.

The twenty-first anniversary year of the Branch was marked by the Committee with Mr. H. G. Parker, one of the Branch's founder-members, as Chairman, selecting a series of lectures which dealt with some of the wide and varied aeronautical developments and activities which have taken place in the years following the foundation of the branch. Attendance at the lectures averaged 70.

The 1960-61 session has had up to the present time two notable lectures. "The S.C.1—an Experiment in Vertical Take-Off" by Mr. David Keith-Lucas was attended by 1,000 persons which, it is thought, is probably a record for a Branch lecture! Secondly, as a Main Society lecture the Branch instituted a named lecture "The Short Brothers' Commemorative Lecture" the first of which was given by Mr. J. L. Parker, formerly Chief Test Pilot and later a Director of Shorts. Mr. J. L. Parker has had a long association with the Branch and indeed gave its second lecture in February 1939.

Over the years successive Chairmen and Committees have given valuable service to the Branch and although due to a period of recession at the moment membership has fallen to 240, efforts are being made to overcome this set-back and climb back to pass the previous highest total membership of 280 and to improve lecture attendances.

First Branch Meeting in The New Lecture Theatre

Sir George Edwards, in his capacity as Weybridge Branch President, welcomed Mr. Keith-Lucas, technical director of Short Brothers and Harland Limited, the lecturer at the first Branch meeting to be held in the parent Society's new lecture theatre at Hamilton Place, on 4th January. The audience of approximately 250 included some 200 Weybridge Branch members who came up by coach, rail and car to attend this notable "first."

The lecture "An Experiment in Vertical Take-Off" matched the occasion—both "experiments" were successful. After a tribute from Sir George to Mr. Keith-Lucas and the team he leads on the Short S.C.1 the lecturer told the development story of the S.C.1 project with, as he said, "no salesmanship if I can avoid it." The tale unfolded in clear and honest fashion with the aid of short pieces of film, slides and an ingenious progressively stabilised see-saw model, to demonstrate the requirement for auto-stabilisation in a VTO aircraft.

The sophisticated lecture presentation and the unaccustomed dignity of the meeting place in no way stifled the lively question period. In fact, so well had Mr. Keith-Lucas "sold" the S.C.1 that one felt, like the new lecture theatre, VTO was here to stay. Technical and monetary considerations might limit initial enthusiasm but this was just the beginning.—G.W.-W.

Graduates' and Students' Section

B.O.A.C. and the Future

On Wednesday 14th December 1960, Mr. Charles Abell, O.B.E., F.R.Ae.S., M.S.L.A.E., Chief Engineer of British Overseas Airways Corporation gave us the benefit of his experience and knowledge regarding the probable direction of B.O.A.C. operations.

He began by explaining that, in common with most other progressive industries, B.O.A.C. have two main plans for the future: (*a*) a *firm* long-term plan, and (*b*) a tentative long, long-term plan (crystal-balling). Both are influenced and progressed by technical development of the equipment and the requirements of the public as these are shown to be necessary and commercially prudent.

Despite confidence in future plans it has to be appreciated that these are subject to violent alteration by events beyond the control of the planner, and Mr. Abell quoted the examples of (*i*) an aircraft that failed to meet its Specification, (*ii*) an aircraft that had to be grounded or withdrawn from service, (*iii*) late delivery of a new type of aircraft, (*iv*) an industrial disturbance, (*v*) an aircraft with too many teething troubles (*vi*) an international situation, and so on. Any of these events could, and do, cost a million pounds, and it may then be necessary to purchase replacement aircraft, re-route services to avoid political and military trouble-spots and then re-cast the whole of the next 5 to 10 year plan.

The organisation of B.O.A.C. was discussed in some detail with particular emphasis on the Engineering and Maintenance Department which Mr. Abell controls.

Among the many slides used to illustrate the lecture was one giving a comprehensive picture of Fleet types and strength for the period 1958 to 1965. Whereas Fleet strength will remain at about the present level until 1965, as a result of the increased size and speed of new types and the increase in frequency the capacity/ton/miles will grow considerably during the same period. Passenger trends, 1952 to 1962, showed a great move from de Luxe and First Class to Tourist and Economy Class. Use of closer pitched seats had enabled the fares to be reduced which, in turn, attracted more passengers. " Low Fares—Full Aircraft—that's the way to make money " was Mr. Abell's observation.

The main objective is to produce an actual load factor, which is as high as possible above the break-even load factor. A gap on the wrong side produced a loss of £4½ million in 1950/51 when the gap was 15 per cent, and a 2½ per cent gap in 1957/58 produced a loss of £2½ million. This indicated that the penalty becomes greater as the operating costs go up and hence there is intensive planning to keep the utilisation of the new expensive monsters at a high level, coupled with short transit and turn-rounds and a persistent campaign to cut down on operating costs. The three watch words of successful operation, engraved on the hearts of all B.O.A.C. staff, are Safety, Punctuality (with regularity) and Economy. The first requirement of Safety demands the highest standards of maintenance, the best staff training, equipment and spares, and the best flying training, flight rules and discipline. Punctuality (with regularity) demands all the previous requirements plus high design integrity in the aircraft, engines and their components. Utilisation is the key to the third requirement of Economy.

In consideration of long, long-term plans Mr. Abell began with taking Supersonics as the first example of crystal ball studies. Current effort is concentrated on the Mach 2·2 aircraft with a seating capacity of around 120 Economy Class passengers, and with a view to its operation on the North Atlantic routes. It was considered that premium fares which may have to apply to the initial stages of operation will stand a better chance of paying off on this " blue riband " route. An aircraft of this type could be flying in seven years. A comparison of the supersonic with the subsonic aircraft gave rise to these interesting figures: (subsonic characteristics in brackets): payload by volume, 28,000 lb. (45 000 lb.), freight hold capacity, 4,000 lb. (13,000 lb.), length, 180 ft. (152 ft.), span, 78 ft. (142 ft.), maximum pressure differential, 12 (8·6), approach speed, 150 knots (140 knots), block speed, 980 knots (435 knots), and cruise altitude, 50-60,000 ft. (40,000 ft.). B.O.A.C. specialists are currently working with the Industry and the Royal Aircraft Establishment on a large number of associated problems, such as layouts, sonic bangs, navigation and auto control, routes and operating costs. It may, of course, happen that the supersonic air liner is not the right answer, particularly if it means higher fares, and there may be a requirement for supersonics and subsonics operating together, each carrying their most lucrative fare class.

The question of freighting was discussed, and to this end B.O.A.C. have converted some of the DC Seven Seas to pure freighters as a step into the market. Dependent on the success of this first venture, is the degree to which B.O.A.C. will participate in this market in the future.

Automatic landing developments were also discussed, and Mr. Abell concluded a fascinating lecture with the assurance that somebody in B.O.A.C. was already looking at the farthest horizon and they would be ready.

Co-option of Committee Member

Following the change of employment of the Chairman, Mr. Wilson, Mr. J. C. Brown (Fairey Division of Westland Aircraft Ltd.) was co-opted to the Section Committee as the representative of the Rotorcraft Section.

Paris Aero Show

Arrangements are in hand for a weekend visit to the Paris Aero Show, which would involve leaving London on Friday 2nd June and returning on Sunday 4th June. Approximate cost is likely to be less than £12, including air and surface fares and hotel accommodation, and as the party must be limited to approximately 35, early application is advised to the Hon. International Visits Secretary, A. H. Fraser Mitchell, 59 Cambridge Road, Kingston-on-Thames, Surrey. Detailed arrangements, deposit required and total cost will be given on this page as soon as they are available.

Test Pilot Lecture

The annual Section Lecture by an eminent test pilot is particularly noteworthy this year. It is to be given by Mr. T. W. Brooke-Smith on " Flight Testing of VTOL Aircraft," at 7.30 p.m. on Wednesday 8th February in the Society's Lecture Theatre.

Annual General Meeting

The Annual General Meeting of the Section will be held on 1st March 1961 **at 8.30 p.m.** In accordance with the new rules for the Section one third of the present Committee will resign and there will be a postal ballot to fill the vacancies. Nomination and ballot forms have been sent out and the names of the successful candidates will be announced at the Annual General Meeting.

Immediately after the meeting a Pan-American film on the Boeing 707 and an Abbott and Costello film entitled " High Fly," will be shown.

Future Visits

The following visits have been arranged for 1961:

Saturday 18th March, National Physical Laboratory at Teddington, Middlesex. (Afternoon.)

Wednesday 19th April, National Gas Turbine Establishment at Pyestock. (Afternoon.)

Wednesday 10th May, Handley Page Ltd. (Afternoon.)

Application for the visits should be made to the Hon. Visits Secretary, N. R. Craddock, 5 Oxleay Road, Harrow, Middlesex (Tel.: Pinner 3633), *at least three weeks before the date of the visit.*

THE LIBRARY

Reviews

INCOMPRESSIBLE AERODYNAMICS. *Edited by Bryan Thwaites. Oxford University Press, London. 1960. 636 pp. Illustrated. 75s.*

This volume is the first of a new series sponsored by the Aeronautical Research Council. Two further volumes are in preparation, one dealing with laminar boundary layers and the other with turbulence: the volume under review covers the steady flow of incompressible fluids about bodies and wings, with some treatment of special topics, the accent being on developments associated with aeronautics. The method of preparation follows that of the earlier *Modern Developments in Fluid Dynamics,* edited by Goldstein, the contributions of individual authors being (presumably) largely rewritten by the editor to preserve balance and continuity; this procedure results in a pleasing uniformity of style and presentation.

Before proceeding further we may briefly survey the contents of the volume. The first chapter deals with general principles, basic equations being stated (generally in vector notation) rather than derived; the complementary roles of inviscid and boundary layer theories are discussed, and boundary layer phenomena described in some detail. The second chapter deals briefly with the calculation of laminar and turbulent boundary layers and the next gives an interesting discussion of theoretical models of real flows. Methods of aerofoil design are then discussed and separated flows treated. Chapter V goes on to deal with the calculation by successive approximations of the effect of the presence of the boundary layer on lift, drag and pitching moment of two-dimensional aerofoils. A chapter is then devoted to boundary layer control. Following this there are two chapters dealing with the flow about three-dimensional wings, the effects of thickness distribution and of lift being treated in turn. Next we have the flow past bodies of revolution, aligned with the stream and at incidence, and in the following chapter the flow past wing-body combinations. A chapter is then devoted to engine aerodynamics, in particular to the actuator disc and radial equilibrium theories of compressor design. The final chapter, Chapter XII, deals with a variety of topics: the jet-flapped wing, the wing with separation bubbles and with vortex sheets springing from the leading edge, the annular wing, cascades, and flows with mainstream vorticity.

It will be recognised from this formidable list of contents that even in a book of this length the treatment must be summary rather than exhaustive and that in a subject where so many of the developments have been mathematical a survey such as this cannot make uniformly easy reading. But the editor has evidently been at pains to set out the developments in such a way that through the major part of the book the non-mathematician can profit from the general description of phenomena and gain an appreciation of the flow models and mathematical procedures which have led to important advances.

The writing is clear and fluent but there are occasional lapses from precision. For example on p. 103 we find the following, referring to the formation of a separation bubble: "When the separated boundary layer has undergone this treatment for a short distance it may feel sufficiently rejuvenated to rejoin the surface." This is a singularly imprecise statement, neglecting altogether the basic mechanism of reattachment, namely the entrainment by the turbulent separated layer of the fluid beneath it. Again, the discussion of the extended Kutta-Joukowski condition would not appear to take us beyond the concise statement that the flow leaves the trailing edge smoothly.

Some excellent flow visualisation studies are included as plates; in particular, Fig. III.5 shows the remarkable correspondence of a calculated and a real flow at low Reynolds number. However, it is doubtful whether the photograph due to Bergh (Fig. I.22) is in fact representative of conditions in normally occurring transition, and as another minor criticism the height of the cylinder shown in the frontispiece is in the region of three boundary layer thicknesses and not, as stated on p. 553, of the same order. In the chapter on boundary layer control there are several omissions and inaccuracies. No mention is made of the suppression of transition due to sweep, nor are the flight results of Pfenninger mentioned, though they provide the most convincing and comprehensive evidence yet obtained of the practicability of boundary layer control for low drag, and were published as long ago as March 1957. Then on p. 243 there is what the reviewer believes to be a direct mis-statement; to his knowledge the claim has never been made that full-chord laminar flow was achieved in Lachmann's flight experiments with either perforations or porous strips, nor has the quoted figure for estimated drag reduction been published elsewhere. In general, this chapter would appear less satisfactory than the remainder of the book, both in arrangement and in content.

It is hoped that the foregoing criticisms will not obscure the very real value of this volume which is, in the main, a lucid and authoritative survey and a worthy successor to "Modern Developments."—M. R. HEAD.

ADVANCES IN APPLIED MECHANICS, Vol. VI. *Edited by H. L. Dryden and Th. von Kármán. Academic Press, New York. 1960. 294 pp. Diagrams. $9.*

This volume is the sixth in the series *Advances in Applied Mechanics,* and contains five survey papers on various problems of fluid mechanics. The topics are not directly related, and it is convenient to discuss each paper separately.

(*i*) *The Theory of Unsteady Laminar Boundary Layers by K. Stewartson.*

After an introductory chapter, the next two chapters are concerned with exact solutions of the Navier-Stokes equations, for both incompressible and compressible flow, in cases where the boundaries move parallel to themselves so that the fluid would be undisturbed if viscous effects were absent. A chapter is then devoted to problems of incompressible flow where the boundaries do not move parallel to themselves, and the fluid is disturbed everywhere from the time at which the relative motion begins. The effects on the boundary layer of fluctuations in the velocity of the body or of irrotational flow round it are then discussed, and the final chapter deals with the unsteady compressible boundary layer, with special reference to shock tubes.

(*ii*) *Boundary-Layer Theory With Dissociation and Ionization by G. Ludwig and M. Heil.*

The first few chapters develop the kinetic theory for a pure diatomic gas in the presence of dissociation and ionisation, and discuss recent work on the collision cross section for the dissociation of molecules by atomic collisions. The boundary layer equations are then developed for a dissociating gas and their solutions are described for flat-plate and stagnation-point flows.

(*iii*) *The Propagation of Shock Waves Along Ducts of Varying Cross Section by W. Chester.*

The discussion is mainly devoted to propagation along ducts consisting of two sections of uniform cross section joined by a transition length. The first solutions described are for the resultant motion when the shock is moving along the downstream section with uniform speed and strength, and the conditions behind the shock are independent of time. The use of Chisnell's theory to examine the non-steady motion of the shock within the region of changing cross section is then discussed and the results

are compared with the results obtained on the basis of Witham's approach. Chisnell's theory is then extended to the propagation of a shock along a duct in which there is a steady flow ahead of the shock.

(*iv*) *Similarity and Equivalence in Compressible Flow by K. Oswatitsch.*

This is the longest article in the volume, and contains a detailed review of similarity considerations for inviscid compressible flow. The material covered includes applications of linear theory and higher approximations, transonic and hypersonic similarity laws, and similarity considerations for unsteady flows and bodies of low aspect ratio.

(*v*) *Kármán Vortex Streets by R. Wille*

This short article includes a review of the stability theory of vortex streets, and of experimental work on their characteristics.

Each of the papers includes an extensive bibliography, and the reproduction of the book conforms with the high standard of previous volumes in the series. It is recommended not only for specialists in the fields covered by the individual articles, but for non-specialist readers requiring summaries of the present state of knowledge on the topics.—D. W. HOLDER.

AERODYNAMIK DES FLUGZEUGES, Vol. 2. *H. Schlichting and E. Truckenbrodt. Springer, Berlin. 1960. 485 pp. Diagrams. DM. 61.50. (In German.)*

This is the second volume of the textbook on the aerodynamic fundamentals of aircraft design by Professor Schlichting and Professor Truckenbrodt. The first volume, reviewed in the JOURNAL in April 1959, dealt with the basic principles of hydrodynamics, gas dynamics and boundary layer theory, and with the details of two-dimensional aerofoil design in incompressible flow. This second volume completes the work with chapters on finite wings in incompressible flow, two-dimensional and finite wings in compressible flow, and isolated fuselages, followed by shorter chapters on wing-fuselage combinations, on fins and tailplanes, and on flaps and control surfaces. An appendix gives rather crude three-view drawings and dimensions of a number of current aeroplanes, with the Ju 52/3m, Me 109 and Spitfire for comparison.

In each chapter (except the first), subsonic and supersonic flows are dealt with, and in the chapter on finite wings an account of the fundamentals of hypersonic flow contributed by Dr. K. Gersten has been interpolated: this would surely have been more appropriate in Volume 1. The authors have thus set out to survey a very large field in a fairly small number of pages, and it is clear that the preparation has involved much work to distil the essential features from a large number of papers (some 330 are referred to in the bibliographies, together with sixty textbooks).

On the whole, the sections on subsonic flow are probably more satisfactory than those which deal with supersonic flows: there are rather too many figures derived from linearised theory for wings with supersonic edges, showing the familiar fictitious cusps and discontinuities. A few experimental results are given which do indicate the illusory nature of the theoretical results, but even after showing in Fig. 8.72 a really spectacular example of the discrepancies between measured and predicted drag coefficients for a delta wing, the authors have used the theoretical results in a comparison between various plan forms in a fashion which implies that they have some practical significance. The supersonic section of the chapter on wing-fuselage combinations is also disappointing, since it is concentrated almost entirely on the combination of a straight wing with a circular cylindrical fuselage; most designers are now prepared to admit that this is not the right shape to make a supersonic aeroplane. Apart from a short account of the sonic area rule, all the modern work on wing-fuselage interference is missing.

Any comprehensive work such as this takes so long to prepare that there must be difficulty in dealing with recent work, and almost every chapter ends with a brief note on some subject which will have to be expanded in a future edition. For example, the subject of slender wings with leading-edge separations deserves more than one page, and so does the jet flap. One would also hope that in a future edition there might be greater emphasis on the design of shapes with desirable aerodynamic characteristics, rather than on evaluation of characteristics of given shapes.

In spite of these criticisms, this is a work of considerable value to any aerodynamicist prepared to read German; and Springer-Verlag have as usual done a first-class job of producing it. The figures are beautifully reproduced, and the type-setting of the mathematics provides a model of clarity which the editors and printers of some recent British works would do well to study.—J. A. BAGLEY.

AN INTRODUCTION TO THE THEORY OF AIRCRAFT STRUCTURES. *D. Williams. Arnold, London, 1960. 448 pp. Diagrams. 60s.*

Dr. Williams' book fills a gap in the literature on aircraft structures in that it provides an excellent introduction to the more advanced aspects of the subject. In this respect it is a pleasant change to read a book which is adequately described by its title.

The book is partly based on weekly lectures given to the post-graduate aeronautics course at Imperial College in the period 1942-1950. Students who attended this course, however, will be interested to know that these lectures only cover approximately half the contents of the book. Extra subjects covered vary from outstanding complete chapters on such matters as pressure cabin structural design, and dynamic loading of aircraft, to additional parts of chapters such as several pages on critical loads for sandwich panels.

It would have been possible to divide the book into two parts, the first on the static problems of the determination of stress distribution and displacement, and the second on dynamic problems such as flutter and gust response. The author does not make this division but simply divides the book into chapters, beginning with Stress Strain and Displacement Relations, and ending with Instability of Motion. It is apparent from even these two chapter headings that the book has a very wide scope.

In addition to the subjects already mentioned there is a complete chapter on the iterative solution of frame problems based on the original idea of Hardy Cross. There are also comprehensive chapters on torsion and bending theory of thin-walled structures and on axial constraint effects arising from the prevention of warping associated with both these effects. It is in this latter theory that one of this reviewer's two minor criticisms of the book can be made, for it fails to point out the essential unity of the axial constraint effects arising from torsion and bending as demonstrated by Argyris. The other criticism is that tension field effects in shear panels is a subject which is not mentioned. Although this subject is rather more empirical than theoretical, a few paragraphs, at least, would have been worth inclusion.

The connecting link between the part dealing with statics and that dealing with dynamics is provided by a chapter concerned with the method of individual displacements. This method of "deriving deflections, stresses, critical loads and natural frequencies of an elastic system by defining the deformation of the system in terms of the individual displacements of a number of chosen stations distributed over it" has received considerable attention in technical reports by the author. This method is a particularly powerful one when used in the solution of dynamic problems which are discussed in one of four later chapters. These chapters form a unique contribution to the literature on aircraft structures, both for the subject matter and the lucidity of the prose. The subject matter of these chapters can be readily assessed from their titles: "Dynamic and Impulsive

Loads," "Dynamic Loading of Aircraft," "Natural Frequencies and Modes of a Complete Aeroplane Structure" and "Instability of Motion." It is only necessary to add that these subjects are fully discussed with a refreshing engineering approach which will commend itself to all its readers.

It is the reviewer's opinion that here is a book on Aircraft Structures which should be of considerable value to both the student, as a textbook, and to the practising engineer, as a reference book, and which is well worth its purchase price.—R. L. WHEELER.

DICTIONARY OF AEROSPACE. *Frank Gaynor. Peter Owen, London, 1960. 260 pp. 35s.*

The busy technical man may take his dictionaries for granted, if he ever troubles to assess them. There are many qualities which go to make a good one, such as comprehensiveness, accuracy, unbiased approach, format and legibility of type. At a rough estimate, this volume contains about 1,800 entries drawn from astronomy, geophysics, chemistry, physics, and all aspects of aero/space engineering. Names of missiles, rockets and space vehicles are included, as also are the names of organisations such as ABMA, NASA, etc. The dictionary appears to be fairly exhaustive in its listing of subject terms. The listing of named vehicles is also fairly thorough, though some omissions were noted. The "Weapons Systems" (under "WS"), however, number only three, namely 117L, 131B and 132. Only the more prominent American organisations have been included, and only the more commonly encountered propellants are listed—there is, for example, no mention of the boranes. Principal tracking systems such as AZUSA and COTAR are given. Some interesting jargon appears. For instance, a "missileer" can apparently fill up a "flight bird," possibly of the "flying stove pipe" variety with "zip fuel" and send it "down the slot," with the approval of the "pad chief." Similarly, during the countdown of a rocket vehicle, various things such as "scrub" or "hold" may happen, and "T-time" is not a rest period, but the moment at the end of the countdown. If the rocket needs some directional control, a "kick-in-the-apogee" may put it right. During flight, it may be subject to "crescelleration," but within an atmosphere deceleration may be achieved with a "flying umbrella."

Unfortunately, an attempt appears to have been made to seek an encyclopedic dictionary form rather than merely a dictionary. The result of this endeavour, applied unevenly and simultaneously with economy of words, has made number of definitions inadequate, some ambiguous, and some misleading. "After body," has no reference to the main use of this term for the aft portion of a body. The "negative G" and "positive G" definitions are incomprehensible.

There are some misleading entries, *e.g.*—"aeroduct" has no reference to recombination ram-jet, a better known term not included. "Aerodynamic vehicle" is for some reason described as unmanned. "L/D ratio" is given as length-diameter ratio only, with no reference to lift/drag which is perhaps the more common use. (Please let us have either L/D or lift-drag ratio but not L/D ratio!)

The accuracy leaves room for improvement in the following examples. The various forms of "ballistic" are not as tidy as they could be. "Boattail" is erroneously described as cylindrical. "MELLERWAGEN" should be MEILERWAGEN. "Mini": here we are told that "minitrack" is a "minitransistor track," which is incomprehensible. Later, under "minitrack," a more appropriate—though brief—definition is given. "NACA" should be ". . . for Aeronautics" *not* ". . . on Aeronautics" also, we are not told that it is now NASA, separately defined later. "Perihelion" here, *aphelion* is also mentioned and is said to be the point nearest to the sun: it is, of course, just the opposite. "Specific impulse"—the definition says it is expressed in pounds per second—should be pounds seconds per pound. "Shrike"—was also a small scale test vehicle for RASCAL; "transsonic"—should be "transonic"; "cabality" should be "capability."

Some words were quite new to me: "afflight"—an approach trajectory that enables the rocket vehicle to survey that portion of the surface of the moon which is turned away from the earth, but without encompassing the moon itself in such trajectory. "Apocynthion"—that position where a rocket or other object is farthest away from the moon. "Cresceleration"—rate of increase of acceleration.

There is a definite American bias, of which the inclusion of American jargon is a good example. Other examples are the missile names (though not space vehicles) which are mainly American, and "ditching device." In the U.K. ditching refers to emergency landing in the sea—the definition does not apply to missile self destruction arrangements.

The apparent hastiness of compilation and the American bias are unfortunate, but the dictionary does include many terms not covered by existing glossaries. Dictionaries are always complementary to each other, and this work may deserve a place on library shelves for this reason, and because of its value as a guide to named missiles and space vehicles. There are other aerospace dictionaries of course, but some of them cost almost £10, so at 35s. this is not desperately expensive. The inclusion of "missilry" jargon can be very useful, and next time we encounter a "cutie pie" we shall know that its main purpose is to measure—not emit—dangerous radiation.—JOHN E. ALLEN.

INTERNATIONAL MISSILE AND SPACECRAFT GUIDE. *Frederick I. Ordway III and Ronald C. Wakeford. McGraw-Hill, London, 1960. 411 pp. Illustrated. £9 14s. 0d.*

Great developments have taken place in the field of missiles and spacecraft since the war and the annual cost of these programmes has reached astronomical proportions running into thousands of millions of pounds. This Guide has been compiled to describe the end-products of this vast development programme, and represents the first attempt to publish an illustrated survey of all missiles and spacecraft throughout the world, of which details have been released.

Although the erratic behaviour of rockets deterred military planners from using them until the Second World War, the literature concerning both pre-war and subsequent developments that has been produced has exceeded all expectations. Much of this literature now lies in comparative isolation and relative seclusion, the information it contains having a very restricted circulation in relation to the total quantity produced. The authors have collected together much of this information and, although unclassified information has been used in the compilation of this Guide, some of the details have only become unclassified while the book was being prepared. For reasons of security this Guide can never be complete, but it makes an extremely valuable contribution to astronautics literature by summarising within one cover all the details of each missile or spacecraft described by thousands of scattered facts, published in Journals all over the world. Some of this information has been taken from military archives direct and it has never been previously released to the Press. Details are also given about little-known missiles which never reached the manufacturing stage, or were later abandoned because their requirements changed. Obviously American missiles have received better treatment in this volume because more information was available than in the case of missiles manufactured in other countries.

The Guide is divided into two main parts: Part 1 deals

with the background to missile and spacecraft progress; Part 2 describes current missiles. The first part contains a history of missile development arranged in chronological order from the year 1200 A.D. together with tables of important vehicle launchings. This is followed by an international survey of the different organisations of missile activities in the countries concerned. Certain early developments have been selected for inclusion in the Guide and these are presented under the following headings: surface-to-surface; surface-to-air; air-to-surface and air-to-air missiles; rocket powered aircraft; sounding rockets; test missiles; drones; special vehicles; manned missiles and spacecraft. Projected space programmes are also described in Part 1. The second part describes all current missiles, the information being presented in the same form of categorisation as used in the first part. This is followed by a section devoted to new developments and, finally, information received too late for inclusion in the body of the work.

One cannot fail to be impressed by this book, if only to appreciate the mammoth task of collecting and collating the information. The high standard of McGraw-Hill has been maintained and the reproduction of copious half-tone and line drawings leaves little to be desired. The only comment the reviewer would wish to make is that the pages are not all consecutively numbered in the usual way, but employ an alphanumerical system for about half the book. —M. D. HULL.

AEROTHERMODYNAMICS AND FLOW IN TURBO-MACHINES. *M. H. Vavra. Chapman and Hall, London, 1960. 609 pp. Diagrams. 116s.*

Dr. Vavra is Professor of the Department of Aeronautics of the U.S. Naval Postgraduate School; his technical education was received in Switzerland under Professor Ackeret. These facts are reflected in the style of his book, which shows experience in teaching and the influence of mid-European ways of thought.

In the 470 pages of his main text, the author does not attempt to cover all aspects of turbo-machines. Some topics are not treated explicitly, or are only briefly touched upon. Dr. Vavra's aim is not to provide a design manual, but to lay a sound theoretical basis upon which detailed investigation of specific problems may rest. He also concentrates upon flow conditions near the design point; stall in particular receives very little discussion.

The reader is expected to be of advanced or post-graduate standard, and to be familiar with the general principles of turbo-machinery and the associated terminology.

The main text is divided into three Parts, of which the first two, entitled "Fundamental Flow Relations" and "Particular Fluid Motions," use vector and dyadic (low-order tensor) analysis almost throughout, to establish with great rigour and clarity fundamental relations for the analysis of arbitrary flows, and of flows of more specifically turbo-machine character, respectively. Seven appendices, covering 84 pages, review all the vector and dyadic algebra required. Of the power and economy of this method there can be no doubt. If it is not as familiar as it should be to many concerned with turbo-machinery, the prospective reader need not be deterred, for the lucidity of the text which accompanies the analysis is so great that the whole can be readily followed with only occasional reference to the appendices.

Part 1 could, indeed, hardly be bettered. Chapter headings are: Kinematics of fluid particles; General stress conditions of fluid particles; Equation of motion; Energy equation; Momentum theorum; Law of moment of momentum. Special attention is given to revealing and explaining the underlying assumptions and hypotheses required. The treatment of the momentum theorum is particularly good, with an analysis of the circumstances in which the enforced neglect of the indeterminable shear stress integrals over the inflow and outflow control surfaces is (and is not) justifiable.

Chapter headings in Part 2 are: Relative motions; Rotational motion; Isentropic flow of perfect gases; Potential flows. A discussion, in this Part, of steady, isentropic, compressible flows with various types of vortex motion in annular passages leads to a particularly striking demonstration of the precautions which must be taken in designing test rigs for measurement of performance of cylindrical cascades. Valuable emphasis is placed on the deductions that the flows in rotating and in stationary blade rows are essentially different, and that in neither case can the flows be axi-symmetrical.

Part 3 is entitled "Applications for turbomachines." Of its six chapters, only two continue the vector treatment. The others deal with velocity triangles but contain little vector algebra. The deductions of Part 2, already noted, imply that the accepted methods of design, based on two-dimensional data, are fundamentally incorrect. Part 3 accepts that this conclusion has to be ignored for want of better data, but continues to point out the resulting un-realities. In the first, very long chapter, axi-symmetric flows are analysed as rigorously as the underlying contradiction permits, a method of calculating such flows in arbitrary cascades is developed, and a comparison is made with experiment on a rotating cascade. A conclusion reached in the course of this analysis runs: "in general, the flow in a stationary cascade cannot be of the free-vortex type. It is for this reason that the necessity of maintaining free-vortex flow in turbo-machines is not even realistic." How many writers of textbooks have failed to recognise this! The next chapter deals with quasi-two-dimensional flows in turbo-machines, and includes a discussion of the vortex systems associated with cascades. Next comes a chapter on axial-flow compressor stages, in which we find the usual velocity-diagram treatment somewhat extended, discussions of blade-loading criteria, Mach number effects, profile drag, and tip losses. Extensive and valuable as the chapter is, one's belief in the adequacy of the two-dimensional axi-symmetric approach has been so thoroughly undermined, that one feels unconvinced that the derivation therefrom of optimum operating conditions is much more than an exercise in algebra, and here, as elsewhere, the proportion of explicit experimental confirmation is small, however much may be implicit in the opinions expressed. The next chapter gives a method of calculating the pressure distribution for potential flow through arbitrary cascades, based on a combination of conformal mapping with data from electric field plotting using conducting paper; again, although the method appears practical and novel, not all readers will find adequate the single comparison with experiment presented. A chapter on one-dimensional flows through turbine stages follows, leading to the presentation of optimum efficiency conditions for a range of design parameters, obtained by the use of a digital computer. Finally, a simplified three-dimensional analysis of axial turbo-machines winds up the main text.

There is a bibliography of more than 100 textbooks and papers, most of which are referred to in the text. The extensive index covers 21 double-column pages.

To sum up, the book must be read by the analyst of flows in turbo-machines, more especially for the rigorous, clear, and enlightening treatment of Parts 1 and 2. Those concerned with cascade tests will find it valuable. The designer will find much useful information in Part 3, but will be conscious of the (deliberate) omissions, and may well feel that the heavy theory of Parts 1 and 2, despite the valuable by-products of

which some examples have been cited, does not yield sufficient positive information of immediate applicability to justify the effort required to master it.—W. J. D. ANNAND.

HYPERSONIC FLOW—Proceedings of the Eleventh Symposium of the Colston Research Society held in the University of Bristol. *A. R. Collar and J. Tinkler (Editors). Butterworths, London, 1960. 432 pp. Illustrated. 70s.*

At a time when one hears it said that there are too many conferences it is refreshing to attend one that is both useful and different. This symposium provided probably the most concentrated three and a half days of hypersonics ever held; all other popular aeronautical topics were expressly excluded such as spaceflight, supersonic aircraft, VTOL or airline economics.

This was an aerodynamic symposium sponsored by a non-aeronautical society, and it was this that contributed very largely to its novelty. The Colston Research Society has held an annual symposium since 1948 on a wide variety of subjects (all non-aeronautical) each selected as being at an active and interesting stage of development. Hypersonics can claim this also, as was shown by the wide scope of the papers presented, of which nearly half were given by overseas visitors.

Of these 15 papers, three dealt with the design problems of hypersonic vehicles, four with theoretical hypersonic aerodynamics and the remaining eight report experimental apparatus, techniques and results. This last selection of papers gives a comprehensive overall picture of the status of experimental facilities for hypersonic research in 1959.

For a body re-entering the earth's atmosphere, the determination of the flow field between the bow shock wave and the body is very important and the papers by M. D. van Dyke, and K. W. Mangler are serious contributions to the solution of this problem. Newtonian flow over a surface is treated by J. P. Guiraud in the third theoretical paper.

B. D. Henshall's paper on the N.P.L. shock tunnel was an elegant piece of work, but the severe limitation of short running time of this equipment for many tasks was brought out by Figure 18. An interesting contrast in scale of cost of experimental facilities is afforded by R. K. Lobb's description of the massive apparatus of the N.O.L. in the U.S.A., and those at A.R.D.E. (R. N. Cox), and Southampton University (K. N. C. Bray *et al.*).

The aeronautical engineer confronted with the design of hypersonic vehicles finds that both theory and experiment, although progressing rapidly, are still some way from providing adequate design information. The two papers by Eggars and Jamison are interesting exercises in the design of hypersonic vehicles. Eggars' work is, of course, well known in the U.K. It was a pleasure to hear him describe the bold manner of approach used at N.A.S.A. Whereas Eggars dealt almost exclusively with winged rocket vehicles for long range, Jamison considered the air-breathing alternative. Metcalf reviewed configurations particularly relevant to hypersonic interceptor missiles. Further thought must be given to the behaviour of hypersonic configurations at other flight speeds; the complete flight paths, a knowledge of the overall stability and control and the experimental confirmation of the estimated design data will be required before the attractive propositions discussed become realities.

The mere written report cannot, however, convey the spirit of the symposium. There was a unanimous opinion that this was a good conference. The audience was invited and consisted mainly of those actively engaged in the field. The arrangements were well organised and the City of Bristol was a fine centre for exploration. One feels that it was the tradition of the Colston Society, which has specialised in arranging this kind of symposium, that gave it its originality. The excellent arrangements made by Professor Collar and Dr. J. Black of the University of Bristol were welcomed by all delegates.

The editors are to be congratulated on publishing the proceedings in a little under nine months, particularly as a well edited but full record of the discussions was also included. The problem of accurately reporting—not so much what contributors say, as what they think—is an increasing chore of conferences. To some, the solution is to omit them. Tape recording was used here with transcripts fed back to speakers weeks after the meeting concluded. One feels that too little attention is given to the winding up of conferences compared with the care which must be taken in their preparation. For example, one might have verbatim reportage supplemented by tape recording and have speakers correct their drafts on the last day. Some time might also be devoted, with advantage, to considering general implications of the subject and suggestions for future papers. The typography of the book is excellent and, for its topical papers and good balance of subject matter, it is highly recommended.—JOHN E. ALLEN.

THE YEARS OF THE SKY KINGS. *Arch Whitehouse. MacDonald, London, 1960. 334 pp. Illustrated. 25s.*

The theme of this book is the air war of 1914-1918. Its author served as an N.C.O. air-gunner in 22 Squadron through 1917 into the beginning of 1918. He complains, with truth, that air gunners were not listed as aces for the kills they made and tells us that when he left 22 Squadron early in 1918 the recording officer told him that Squadron records credited him with 16 enemy aircraft destroyed and six kite balloons.

In his foreword he writes: "I was born with a pack rat's instinct to collect data, details, photographs, newspaper clippings, and the few reliable books that were published at the time [and] continued this interest in military aviation and the history of the flyers who fought on either side of the line. I have examined official records in Washington, London and Berlin. When the opportunity rose to prepare this volume, I did not have to resort to the wide research less fortunate writers have found so necessary."

He continues that he has been writing air stories of the First World War and factual accounts of aces and Squadrons for more than 30 years and all the material in his files "had only to be rewritten or incorporated into its proper category or chronological sequence."

I opened the book with pleasant expectation. I found some excellent narratives about the flyers of that war, although the author is careful to point out that some at least are in parts conjectural; all are written in racy, slangy, easy-to-read English. The atmosphere of the period is invoked, even if the author's prejudices lead him to make some astonishing statements, such as: "the R.F.C. actually started for France on August 13—nine days after war was declared. Henderson's chief of staff was Acting Colonel F. Sykes, a Boer War veteran who had been floating about in free balloons since 1904. In 1911 he took his ticket on a Bristol biplane—presumably the early Bristol Bullet." Language of this kind is a disservice to the R.F.C. Moreover, if the author *had* taken the trouble to widen his research he would have learned from Sir Frederick Sykes' autobiography that he took his certificate No. 96 on a 50 h.p. Bristol Boxkite. So why presume anything different?

If the author *had* taken the trouble to make the research he disowns he would have avoided many mistakes and have produced a book that would not have jarred on one

by all too frequent errors of fact and of spelling. I enumerate some below:

Errors of fact: Speeds of the Maurice Farman Longhorn and Shorthorn were not 85 and 95 m.p.h., but about 25 and 30 m.p.h. less. Samson was not responsible for the 1917 Friedrichshafen raid; it was organised by Pemberton-Billing from the U.K. and Belfort. The French had roundels before the British, not the other way about. The first 1½-Strutters did not go to the R.N.A.S. in the autumn of 1915; the prototype was not flown and passed until December 1915. Kauper, of the Kauper gear, was with Sopwiths before the war. No. 6 squadron flew oversea on 7th October 1914, not "On the declaration of war" as Whitehouse writes. On p. 69 Whitehouse writes of a 250 round belt of ammunition for the Vickers gun and on p. 111 of a single belt of 500 rounds!

He writes that men were enticed from the trenches by the lure of 6s. a day flight pay to be gunner observers; but, in fact, 2nd class A/Ms only got 2s. a day flight pay and many flew in that humble rank. Whitehouse writes that twin synchronised guns were standardised in both German and Allied aircraft after the appearance of the D.1 Albatros scout; this was not so, what about the S.E.5(a), most used British scout next to the Camel, with its Foster-mounted fixed-free Lewis? or the twin-Lewis top mounted Camels of the R.N.A.S.? The Fokker Dr.1 Triplane was not "an out-and-out copy of the Sopwith Triplane."

Whitehouse writes that "There is some question whether the Sopwith triplane had as yet been sent to the front [on 7th May 1917]." There is no question about that at all. The prototype did not come out until 28th May 1917 and did not go oversea for Service test until about mid-June.

The author gives us a graphic account of how he was not shot down by Richthofen on 13th April 1917, but by A.A., and he avers that the FE2b flown by a Captain Bush, with Whitehouse as gunner, was wrongly claimed by Richthofen as having been shot down by him at Monchy on that date. This is an interesting story and one would like to hear what General Bodenschatz (then Richtofen's adjutant) has to say. Staffel 11 was then at Douai aerodrome, not Marcke, as Whitehouse believes.

Where Whitehouse refers to a 330 lb. British bomb coming into use in 1916 I think he must mean the 230 lb.

Spelling errors: Verhers for Verkers; Harvey-Kelley for Harvey-Kelly; Sipps for Sippe; Schilling for Schillig (Roads); *Pour le mérit* for *Pour le Mérite;* D.W.F. for D.F.W.; Izelle-Hammeau for Izel-le-Hameau.

Flyers were not gazetted, but posted, to a squadron.

And I can assure Arch Whitehouse that Major W. G. Barker, V.C., did not shoot down the Austrian ace, Oberleutnant Linke-Crawford; that Capt. C. E. Howell was Australian, not British, and that his score was 19 not 30.

If the author will now take the time to check all his facts and can succeed in getting his publishers to produce a revised edition he will have produced a readable and enjoyable book free from the irritations of the too frequent inaccuracies of the present edition.—NORMAN MACMILLAN.

RADIO AIDS TO CIVIL AVIATION. *R. F. Hansford (Editor). Heywood, London. 1960. 623 pp. Illustrated. 126s.*

The rapid expansion of Civil Aviation and the increased performance of modern aircraft give rise to many problems. To keep pace with them it is essential that the maximum possible use be made of the increased capabilities of modern radio aids. This book is written particularly for the operational planner and system designer actively engaged in planning and developing future systems to meet the new requirements. It aims to provide a comprehensive survey of both the operational problems and the basic information relating to radio aids.

The contents are grouped under nine headings: Introduction; Principles of Radio; Aircraft Communication Systems; En route Aids; Aids to Traffic Control; Aids to Approach, Landing and Surface Movement; Weather Radar; Radar Links and Present Trends. They have been written by the Editor and fourteen collaborating authors, each an expert in his own particular subject.

Principles of Radio deals with those topics necessary for a fuller understanding of the subsequent chapters and covers propagation, together with the effects of the earth's surface and the ionosphere, the basic principles of bearing measurement, and the use of hyperbolic systems for position fixing. There is also a short section on noise in which one of the worked examples unfortunately proclaims that $4 \times 5 = 15$.

Aircraft Communication Systems describes long and short range R.T. systems and includes some information theory and a discussion on the relative merits of plain speech and code. An interesting table is given comparing Morse's original code with that now internationally used, though someone has inadvertently invented a new symbol for X.

The next three chapters give a comprehensive description of radio aids for Navigation, Traffic Control, Approach, Landing and Surface Movement and include a large number of well-known systems. Weather Radar deals with both ground and airborne equipment and includes a section on the types of meteorological conditions which can be detected. Radar Links are also discussed since they are of importance when diverse information is required at one central control point. The final chapter is entitled Present Trends and examines the problems of the future and some of the likely developments.

There are two appendices, one giving some relevant I.C.A.O. regulations, and the other a summary of Equipment Characteristics. There is also a Glossary of useful terms and an index which, together with the five page list of contents, assists rapid access to any particular topic.

This is a well printed book and the quality of the many diagrams and photographs is excellent. The balance between the description of equipments and the technical principles involved has been well chosen, and for those who require more detailed treatments each chapter has a bibliography.

It was the Editor's intention that this book should be a work of reference. He has achieved more than this. Not only in the final chapter, but all through the book, the limitations of present equipment and the need for future developments are constantly stressed. System designers and planners should find much in this book to stimulate their efforts in solving the future problems of Civil Aviation.—D. H. PAUL.

Additions to the Library

Axiomatics of Classical Statistical Mechanics. R. Kurth. Pergamon, Oxford. 1960. 180 pp. 45s. ". . . . an attempt to construct classical statistical mechanics as a deductive system, founded only on the equation of motion and a few well-known postulates which formally describe the concept of probability." The major part consists of theorems and proofs and the chapter headings are Introduction, Mathematical Tools, The Phase Flows of Mechanical Systems, The Initial Distribution of Probability in the Phase Space, Probability Distributions which Depend on Time, Time-Independent Probability Distributions, Statistical Thermodynamics.

Electrical Noise. W. R. Bennett. McGraw-Hill, New York. 1960. 280 pp. Diagrams. 77s. 6d. *To be reviewed.*

Electrodynamics of Continuous Media. L. D. Landau and E. M. Lifshitz. Pergamon, London. 1960. 417 pp. 84s. Volume VIII of the *Course of Theoretical Physics*, translated from the Russian by J. B. Sykes and J. S. Bell; five volumes have already appeared. This one deals with the theory of electromagnetic fields in matter and with the theory of the macroscopic electric and magnetic properties of matter; it includes magnetic fluid dynamics and diffraction of X-rays in crystals.

Field Theory of Guided Waves. R. E. Collin. McGraw-Hill, New York. 1960. 606 pp. Diagrams. 128s. An account of guided wave theory at the graduate level, setting out the mathematical techniques and solutions of the more elaborate waveguiding structures, waveguide discontinuities, waveguide antennae and coupling apertures. Based on lecture notes for a graduate course in guided wave theory given at Laval University and work done at the Case Institute of Technology.

F. W. Lanchester: A Life of an Engineer. P. W. Kingsford. Edward Arnold, London. 1960. 246 pp. Illustrated. 30s. *To be reviewed*

Inertial Guidance. C. S. Draper *et al.* Pergamon, Oxford. 1960. 130 pp. Illustrated. 42s. *To be reviewed.*

Nuclear Propulsion. M. W. Thring. Butterworth, London. 1960. 300 pp. Diagrams. 50s. *To be reviewed.*

Solid Propellant Rocket Research. M. Summerfield. Academic Press, New York. 1960. 692 pp. Illustrated. $6.50. *To be reviewed.*

Space Trajectories. Technical Staff, Radiation Inc. Academic Press, New York. 1960. 298 pp. Diagrams. $12. *To be reviewed.*

Spitfire—The Story of a Famous Fighter. B. Robertson. Harleyford Publications, Letchworth. 1960. 211 pp. Illustrated. 45s. *To be reviewed.*

Thermodynamic and Transport Properties of Gases, Liquids and Solids. Papers presented at Symposium on Thermal Properties. Y. S. Touloukian, General Chairman. A.S.M.E. and McGraw-Hill. 472 pp. Diagrams. 97s. Contains the papers presented at the Symposium on Thermal Properties sponsored by the Standing Committee on Thermophysical Properties of the Heat Transfer Division of the American Society of Mechanical Engineers, held in February 1959 at Purdue University, Lafayette, Indiana, contributions being received from Germany, Switzerland, the U.K. and U.S.A. The papers are divided into nine groups: (1) Theoretical Estimation of Transport Properties. (2) Review of Recent Work on Transport Properties. (3) Thermodynamic Properties: Gases and Liquids. (4) PVT Data and Equation of State. (5) Thermodynamic Properties of Boron Compounds. (6 and 7) Transport Properties: Experimental. (8) High Temperature Transport Properties of Metals and Ceramics. (9) High Temperature Thermodynamic Properties of Gases. While some papers are not closely connected with aeronautics (Measurements of Thermal Conductivity of Milk) the greater part contains papers of interest to the high-temperature gas dynamicist, and the applied physicist. Those concerned with newer plants and fuels, and the structural designer, will also find useful information.

Reports

AERODYNAMICS

BOUNDARY LAYER *see also* AIRCRAFT OPERATION

Flight investigations of the low-speed characteristics of a 45° swept-wing fighter-type airplane with blowing boundary-layer control applied to the leading- and trailing-edge flaps. H. Quigley et al. *N.A.S.A. T.N. D-321. Sept.* 1960.
Includes documentation of the low-speed handling qualities as well as the pilots' evaluation of the landing-approach characteristics. The results are compared with those for the aeroplane with a slatted leading edge and the same trailing-edge flap. Several of the factors limiting the use of the high lift possible with leading-edge boundary layer control flaps were associated with the high angles of attack required for low-speed flight in the test aeroplane.—(1.1.6.1×5×3.6×1.3.4×1.8.0.1).

Photographic evidence of streamwise arrays of vortices in boundary-layer flow. E. J. Hopkins et al. *N.A.S.A. T.N. D-328. Sept.* 1960.
Photographs are presented of various models coated with fluorescent oil to show evidence of surface vortices at a Mach number of 3·03.—(1.1.0.4×1.4).

Radiation shielding of the stagnation region by transpiration of an opaque gas. J. T. Howe. N.A.S.A. T.N. D-329. Sept. 1960.
The laminar compressible boundary layer in the two-dimensional and axisymmetric stagnation regions has been analysed to show the effects of the injection of a radiation-absorbing foreign gas on an incident radiation field, and on the enthalpy profiles across the boundary layer. Total heat transfer to the stagnation region is evaluated for numerous cases and the results compared with the no shielding case. Required absorption properties of the foreign gas are determined and compared with properties of known gases.—(1.1.1.4×1.9.1×34.1×25.2).

Investigation of effects of roughness, surface cooling, and shock impingement on boundary-layer transition on a two-dimensional wing. K. R. Czarnecki and J. R. Sevier. N.A.S.A. T.N. D-417. June 1960.
The wing had a sharp leading edge with a flat surface on one side and a constant favourable-pressure-gradient surface on the other. Tests were made at $M=1{\cdot}61$ and 2·01, over a Reynolds number per foot range from about $0{\cdot}5\times10^6$ to approximately $9{\cdot}5\times10^6$. A comparison is made of the lateral spread of turbulence as determined by a total pressure probe technique and as determined by a hot-wire technique.—(1.1.2.4×1.9.1).

COMPRESSIBLE FLOW *see also* STABILITY AND CONTROL
TESTING AND INSTRUMENTS
THERMODYNAMICS
MISSILES

Recombination of oxygen in a nozzle: comparison with experimental results for air from General Electric Research Laboratory. K. N. C. Bray and B. Makin. U.S.A.A. Report 143. *Sept.* 1960.—(1.2.3.1×1.5.1×32.2).

On rotational supersonic flow past thick airfoils—Tables. A. Kogan and A. A. Betser. Israel Inst. of Tech. Report 11. *June* 1960.
Certain shock wave parameters, which appear in the calculation of pressure distribution on aerofoils in a supersonic rotational flow, are presented in tabular form. The method underlying the use of the tables is summarised and discussed. A numerical example is worked out, illustrating the procedure in the application of the tables.—(1.2.3.2×1.10.1.1).

CONTROLS *see also* BOUNDARY LAYER
FLUID DYNAMICS
AEROELASTICITY

Low-speed investigation of a full-span internal-flow jet-augmented flap on a high-wing model with a 35° swept wing of aspect ratio 7·0. T. R. Turner. N.A.S.A. T.N. D-434. Aug. 1960.
An investigation of full-span internal-flow jet-augmented flap on a semispan jet transport model has been made in the Langley 300 m.p.h. 7×10 ft. tunnel. The large diving moments were trimmed by blowing over the deflected elevator or by downward blowing from a fuselage nose jet. The jet momentum coefficient range was 0 to 6·33, the flap deflection range was 0° to 7°, and the angle-of-attack range extended from −25° to the stall angle.—(1.3.4×1.8.2.2).

FLUID DYNAMICS *see also* BOUNDARY LAYER
AIRCRAFT OPERATION
POWER PLANTS

N.A.S.A. research on the hydrodynamics of the gaseous vortex reactor. R. G. Ragsdale. N.A.S.A. T.N. D-288. Sept. 1960.
A gaseous nuclear reactor utilising a vortex-type flow field to increase the heavy-element holdup time is considered. Equations describing the separation mechanism are presented for both ideal (inviscid, laminar) and real (viscous, turbulent) fluid conditions. Analytical results are given for the separation of air-bromine mixtures in a vortex test apparatus. The programme is described and some data are shown. (1.4×27.6).

Investigation of the flow in a rectangular cavity in a flat plate at a Mach number of 3·55. R. W. McDearmon. N.A.S.A. T.N. D-523. Sept. 1960.
An investigation has been made in the Mach 3·5 blow-down jet of the Langley high-temperature fluid mechanics section of the effects of systematic variations of depth, span, and upstream and downstream lip radii on the flow phenomena in a rectangular cavity in a flat plate. Pressure distributions were measured and schlieren photographs of the shock structure immediately above the cavity were taken.—(1.4×1.6.1×1.3).

INTERNAL FLOW *see also* COMPRESSIBLE FLOW
TESTING AND INSTRUMENTS
AIRCRAFT OPERATION

Investigations on an experimental air-cooled turbine. Part III —The effects of cooling on the overall turbine aerodynamic performance and initial operation at an inlet gas temperature of 1400° K. D. E. Fray and N. E. Waldren. Part IV—Examination of sintered vitallium air-cooled turbine blades following 100 hours' operation at a turbine inlet mean gas temperature of 1400° K. N. E. Waldren and C. J. Hart. R. & M. 3144. 1960.—(1.5.3.1×34.3.4×27.1×33.2.4.1.9).

LOADS *see* FLUID DYNAMICS
STABILITY AND CONTROL
AEROELASTICITY
AIRCRAFT
MISSILES

PERFORMANCE *see* STABILITY AND CONTROL
AIRCRAFT

STABILITY AND CONTROL *see also* BOUNDARY LAYER
CONTROLS
WINGS AND AEROFOILS
AIRCRAFT
AIRCRAFT OPERATION
EXTRA-ATMOSPHERIC
TECHNOLOGY
MISSILES

Wind tunnel measurements of normal force and pitching moment on four cone-cylinder combinations at transonic and supersonic speeds. E. Huntley. C.P. 507. 1960.
Wind tunnel tests were made, in the 3×3 ft. tunnel at R.A.E. Bedford, on four cone-cylinder models at Mach numbers between 0·70 and 2·00. The tip of each cone was rounded and the overall fineness of each model was less than 8·0.—(1.8.2.2 ×1.2.2×1.2.3).

*Flight investigation of an automatic pitchup control. G. J. Hurt and J. B. Whitten. N.A.S.A. T.N. D-*114. *Aug.* 1960.—(1.8.2.2×13.2).

*An analytical method for studying the lateral motion of atmosphere entry vehicles. R. E. Slye. N.A.S.A. T.N. D-*325. *Sept.* 1960.
The method is applicable for small entry angles and results obtained with the method are compared with results from numerical integration of the complete equations of motion. The method is found useful for studying manoeuvres of vehicles with aerodynamic lift-drag ratios up to about 1·5.—(1.8.1.1× 8.2×25.1).

*Theoretical calculations of the pressures, forces, and moments due to various lateral motions acting on tapered sweptback vertical tails with supersonic leading and trailing edges. K. Margolis and M. H. Elliott. N.A.S.A. T.N. D-*383. *Aug.* 1960.
Expressions based on linearised thin aerofoil theory for supersonic speeds are derived for constant sideslip, steady rolling, steady yawing, and constant lateral acceleration. Design-type charts are presented which enable estimation of stability derivatives for given values of Mach number and tail-geometry parameters. All results are, in general, applicable at those supersonic speeds for which both the tail leading and trailing edges are supersonic.—(1.8.1.2×1.6.1).

*Effect of Reynolds number on the lateral-stability derivatives at low speed of sweptback- and delta-wing-fuselage combinations oscillating in yaw. C. P. Llewellyn and W. D. Wolhart. N.A.S.A. T.N. D-*398. *Aug.* 1960.
Results were obtained from the models oscillating in yaw over an angle-of-attack range from 0° to 32° for the delta wing models and from 0° to 28° for the swept-back wing model. The Reynolds number range was from $0{\cdot}7\times10^6$ to 7×10^6 for the swept-back wing model and from $0{\cdot}9\times10^6$ to 9×10^6 for the delta wing models. Amplitudes of oscillation were from 2° to 10° and reduced-frequency parameters from 0·028 to 0·113.—(1.8.1.2).

*Aerodynamic performance and static stability at Mach number 3·3 of an aircraft configuration employing three triangular wing panels and a body of equal length. C. S. James. N.A.S.A. T.N. D-*330. *Aug.* 1960.
Wind tunnel measurements of performance and static stability at combined angles of attack and sideslip were made on a supersonic aircraft configuration at a Mach number of 3·3 and a Reynolds number of 5·46 million.—(1.8.0.2×1.7).

*Theoretical analysis of the longitudinal behaviour of an automatically controlled supersonic interceptor during the attack phase against manœuvering and nonmanœuvering targets. C. H. Woodling and O. B. Gates. N.A.S.A. T.N. D-*454. *June* 1960.
First-order guidance is utilised and the interceptor flight path is controlled by either a pitch-rate or a normal-acceleration command system. A discussion is given of the system bias errors due to first-order guidance or speed changes, and methods of minimising or eliminating these errors are presented. The effect of control rate limiting is also discussed.—(1.8.2.1×5).

THERMO-AERODYNAMICS *see also* BOUNDARY LAYER
AEROELASTICITY
EXTRA-ATMOSPHERIC
TECHNOLOGY

Motion and heating during atmosphere reentry of space vehicles. T. J. Wong et al. *N.A.S.A. T.N. D-*334. *Sept.* 1960. —(1.9.1×8.2×25.3).

WINGS AND AEROFOILS *see also* COMPRESSIBLE FLOW
AEROELASTICITY

Lift and drag of single wedge sections in two dimensional transonic flow. J. W. Bugler and N. C. Hanslip. C.o.A. Report 131. *May* 1960.—(1.10.2.2×1.12.1×17.1).

Supersonic wings of minimum drag having a given static margin. E. V. Bulygina. R.A.E. Lib. Trans. 912. *Sept.* 1960.
The problem considered is that of the design of a lifting surface for flight at a supersonic speed which has a static margin large enough to permit stable flight at subsonic speeds. The usual assumptions of linearised theory are made. The plan forms considered have straight, unswept trailing edges and leading edges which are supersonic everywhere (implying the absence of side edges). Lengthwise camber is applied to obtain the required zero-lift pitching moment.—(1.10.1.2×1.8.2.1).

*A wind-tunnel investigation of the development of lift on wings in accelerated longitudinal motion. T. R. Turner. N.A.S.A. T.N. D-*422. *Aug.* 1960.
Results are presented of a wind tunnel investigation to determine the effect of acceleration on the development of lift in a simulated catapult take-off for a two-dimensional wing, an unswept wing, a swept wing, and a delta plan form wing.—(1.10.2.2×5).

HELICOPTER AERODYNAMICS

*Effect of blade cutout on power required by helicopters operating at high tip-speed ratios. A. Gessow and F. B. Gustafson. N.A.S.A. T.N. D-*382. *Sept.* 1960.
A numerical study was made of the effects of blade cutout (varying from 0 to 0·5 of the rotor radius) on the power required by a simple helicopter rotor travelling at tip speed ratios of 0·3, 0·4 and 0·5.—(1.11.3).

*Measurements of the time-averaged and instantaneous induced velocities in the wake of a helicopter rotor hovering at high tip speeds. H. H. Heyson. N.A.S.A. T.N. D-*393. *July* 1960. —(1.11.3).

*Equations for the induced velocities near a lifting rotor with nonuniform azimuthwise vorticity distribution. H. H. Heyson. N.A.S.A. T.N. D-*394. *Aug.* 1960.
Equations are developed for all three induced velocity components at an arbitrary point and for an arbitrary harmonic of the vorticity distribution.—(1.11.3).

*Experimental smoke and electromagnetic analog study of induced flow field about a model rotor in steady flight within ground effect. R. B. Gray. N.A.S.A. T.N. D-*458. *Aug.* 1960. —(1.11.3).

TESTING AND INSTRUMENTS *see also* WINGS AND AEROFOILS
THERMODYNAMICS

A report of work done and visits made in Canada and U.S.A. from Oct. 1959-*Jan.* 1960. *J. Pennelegion. U.S.S.A. Report* 141.—(1.12×11×1.2).

*Theoretical determination of the power efficiency and overall flow behaviour of free jet wind tunnels with special emphasis on transonic wind tunnels. F. van der Walle. N.L.L.-T.R. G.*1. *Dec.* 1957.
The power efficiency and the overall flow characteristics are theoretically derived for a free jet transonic wind tunnel. The formulae appear to be applicable to all types of wind tunnels (sub-, trans- and supersonic; with and without free jets). The results are confirmed by the available experimental data. The power efficiency and overall flow characteristics can be calculated with reasonable accuracy.—(1.12.1.2×1.5.1.4).

AEROELASTICITY

See also STRUCTURES—LOADS
THEORY AND ANALYSIS

Subsonic kernel-function flutter analysis of a highly tapered tail surface and comparison with experimental results. G. D. Walberg. N.A.S.A. T.N. D-379. *Sept.* 1960.
A flutter analysis employing the kernel function for three-dimensional, subsonic, compressible flow is applied to a flutter-tested tail surface which has an aspect ratio of 3·5, a taper ratio of 0·15, and a leading-edge sweepback of 30°. Theoretical and experimental results are compared at Mach numbers from 0·75 to 0·98.—(2×1.6.3).

Investigation at transonic speeds of loading over a 30° *swept-back wing of aspect ratio* 3, *taper ratio* 0·2 *and N.A.C.A.* 65*A*004 *airfoil section mounted on a body. D. D. Arabian. N.A.S.A. T.N. D*-421. *June* 1960.
Two wings were tested, one of solid steel, the other of plastic with a steel core. The twist distributions for the wings were calculated from experimental influence coefficients and pressure data. Studies of flow in the boundary layer are presented.—(2×1.6.1×33.1.1).

Effect of aerodynamic heating on the flutter of a rectangular wing at a Mach number of 2. *H. L. Runyan and N. H. Jones. N.A.S.A. T.N. D*-460. *June* 1960.
Results are presented of an experimental and theoretical investigation of the flutter of a solid aluminium-alloy wing as affected by aerodynamic heating. Good correlation with theory was found.—(2×1.9.1×1.10.2.2×1.3.1.1).

AIRCRAFT

See also AERODYNAMICS—BOUNDARY LAYER
AIRCRAFT OPERATION
EXTRA-ATMOSPHERIC TECHNOLOGY
STRUCTURES—THEORY AND ANALYSIS

An examination of handling qualities criteria for V./S.T.O.L. aircraft. S. B. Anderson. N.A.S.A. T.N. D-331. *July* 1960.
A study has been made of handling qualities of V./S.T.O.L. aircraft. With the current military requirements for aeroplanes and helicopters as a framework, modifications and additions have been made for conversion to V./S.T.O.L. requirements using flight results and pilots' comments from a limited number of V./S.T.O.L. aircraft, B.L.C. equipped aircraft, and flight simulators. The reasoning behind suggested V./S.T.O.L. requirements and the areas where existing information is inadequate and further research is required are discussed.—(3.12×1.7×1.8.0.1×5).

Aerodynamic characteristics of a 1/4*-scale model of a tilt-wing V.T.O.L. aircraft at high angles of wing incidence. L. P. Tosti. N.A.S.A. T.N. D*-390. *Sept.* 1960.
The model had two propellers with hinged (flapping) blades mounted on the wing which could be tilted from 4° incidence for forward flight to 86° for hovering flight. The investigation included measurements of longitudinal stability characteristics for the low-speed portions of the transition range from 60° to 84° wing incidence at zero acceleration (steady level flight) and of lateral stability and control characteristics for wing incidences from 60° to 80° for a range of conditions simulating zero acceleration, 0·25*g* forward acceleration, and 0·25*g* deceleration.—(3.2×1.8.0.2).

Data from a static-thrust investigation of a large-scale general research V.T.O.L.-S.T.O.L. model in ground effect. R. J. Huston and M. M. Winston. N.A.S.A. T.N. D-397. *Aug.* 1960.
Model force data and pressure distribution on wing and flaps at centre of wing semispan, measured at two ground heights above the ground, are presented without analysis for various combinations of propeller disc loading, wing tilt, and flap deflection.—(3.12×1.6.1).

Wind-tunnel investigation of a small-scale model of an aerial vehicle supported by tilting ducted fans. C. C. Smith. N.A.S.A. T.N. D-409. *Aug.* 1960.
The model had three ducted fans arranged in a triangular fashion (one fan at the front and two at the rear) that could be tilted relative to the airframe for forward flight. The results of tests to determine the static longitudinal and lateral stability characteristics of the model are given and a comparison with a tandem fixed-duct configuration is included.—(3.12×1.8.0.2).

Effect of ground proximity on aerodynamic characteristics of two horizontal-attitude jet vertical-take-off-and-landing airplane models. W. A. Newsom. N.A.S.A. T.N. D-419. *Aug.* 1960.
The first model had a tilting wing-engine assembly which was set at 90° incidence for take-off and landing. The second had a cascade of retractable turning vanes to deflect the exhaust of the horizontally mounted jet engines downward for vertical take-off and landing while the entire model remained in a horizontal attitude. Tests were made for a range of heights above the ground and included force tests and tuft studies of the flow field caused by the jet exhaust.—(3.12×1.8.0.1).

AIRCRAFT OPERATION

See also AERODYNAMICS—BOUNDARY LAYER
STABILITY AND CONTROL
WINGS AND AEROFOILS
AIRCRAFT
NAVIGATION

*Tire-to-surface friction-coefficient measurements with a C-*123*B airplane on various runway surfaces. R. H. Sawyer and J. J. Kolnick. N.A.S.A. T.R. R*-20. 1959.
An investigation was conducted to obtain information on the tyre-to-surface friction coefficients available in aircraft braking during the landing run. The tests were made with a C-123B aeroplane on both wet and dry concrete and bituminous pavements and on snow-covered and ice surfaces at speeds from 12 to 115 knots. Measurements were made of the maximum (incipient skidding) friction coefficient, the full-skidding (locked wheel) friction coefficient and the wheel slip ratio during braking.—(5.3×33.1.2×3.6).

Turbulence studies of a rectangular slotted noise-suppressor nozzle. J. C. Laurence. N.A.S.A. T.N. D-294. *Sept.* 1960.
Turbulence studies, made in a model noise-suppressor-type nozzle, were aimed at the statistical properties of the fluctuating flow which may cause the attenuation of the sound power in some frequency bands and at the effect of the nozzle spacing on these properties. Mean and turbulent velocity profiles, intensity, scale, spectra, and probability densities of the fluctuating velocities and convection velocities of the pressure and velocity eddies are reported.—(5.6×1.4.2).

Measurement of screen-size effects on intensity, scale, and spectrum of turbulence in a free subsonic jet. C. D. Howard and J. C. Laurence. N.A.S.A. T.N. D-297. *Aug.* 1960.—(5.6×1.5.1.3).

A flight study of a power-off landing technique applicable to reentry vehicles. R. S. Bray et al. *N.A.S.A. T.N. D*-323. *July* 1960.—(5×8.2×1.8.0.1).

Transonic performance characteristics of several jet noise suppressors. J. W. Schmeer et al. *N.A.S.A. T.N. D*-388. *July* 1960.
An investigation was made of the transonic aerodynamic performance characteristics of six noise-suppressor nozzles using a hot-jet exhaust from a hydrogen peroxide system.—(5.6×27.1).

Some landing studies pertinent to glider-reentry vehicles. J. C. Houbolt and S. A. Batterson. N.A.S.A. T.N. D-448. *Aug.* 1960.
The effect of the initial conditions of sinking velocity, angle of attack, and pitch rate on impact severity and the effect of locating the rear gear in various positions are discussed. Some information is included regarding the influence of landing gear location on effective masses. Preliminary experimental results on the slide-out phase of landing include sliding and rolling friction coefficients that have been determined from tests of various skids and all-metal wheels.—(5×8.2×33.1.2).

Noise considerations for manned reentry vehicles. D. A. Hilton et al. *N.A.S.A. T.N. D*-450. *Sept.* 1960.
Noise measurements pertaining mainly to the static firing, launch, and exit flight phases are presented for three rocket-powered vehicles in the Project Mercury test programme. Both internal and external data from onboard recordings are presented for a range of Mach numbers and dynamic pressures and for different external vehicle shapes.—(5.6×25.2×8.2).

Some measurements of noise transmission and stress response of a 0·020-inch duralumin panel in the presence of air flow. G. T. Kantarges. N.A.S.A. T.N. D-459. *Sept.* 1960.
The noise reduction through the test panel measured in an absorbent chamber is in general agreement with values predicted by the theoretical weight law for the flow-not-attached case. Corresponding data for the flow-attached case do not follow the weight law but rather indicate less noise reduction at the high frequencies.—(5.6×33.1.1×1.1).

Response of a light airplane to roughness of unpaved runways. G. J. Morris and J. W. Stickle. N.A.S.A. T.N. D-510. *Sept.* 1960.
The response of a light liaison-type aeroplane to roughness of unpaved runways has been studied by measuring centre-of-gravity acceleration as the aeroplane was taxied at constant speeds over two grass runways and a concrete taxiway. The results are presented in the form of acceleration distributions, acceleration power spectra, transfer functions, and average peak accelerations as functions of taxying speed. Measurements from elevation profile surveys of the runways and taxiways are given as elevation profiles and power spectra.—(5×3.1×33.1.2).

EXTRA-ATMOSPHERIC TECHNOLOGY

See also AERODYNAMICS—STABILITY AND CONTROL
THERMO-AERODYNAMICS
AIRCRAFT OPERATION
MISSILES
NAVIGATION
POWER PLANTS
THERMODYNAMICS

Description of the experiments performed during the flight of the second Russian space-ship satellite. Pravda 4th-6th Sept. 1960. *R.A.E. Lib. Trans.* 921. *Sept.* 1960.
Extracts from *Pravda* of 4th-6th Sept. 1960, describing the second Russian space-ship satellite, otherwise known as Sputnik 5 (1960 λ), which brought back two dogs and other living animals from orbit. The topics include the orbit and descent path of the satellite, air conditioning in the cabin, the television apparatus, full details of the biological investigations performed, cosmic-ray experiments and measurements of solar X-rays and ultra-violet radiation.—(8.2×8.1×9×32.2).

Ignition of a combustible atmosphere by incandescent carbon wear particles. D. H. Buckley et al. *N.A.S.A. D*-289. *Sept.* 1960.
An investigation was conducted to determine whether carbon wear particles abraded from rotating carbon elements in contact with a metal surface could ignite a combustible gas atmosphere. Experiments were run with a carbon vane in sliding contact with a rotating 440-C stainless steel flat disc in a chamber filled with a combustible atmosphere of air and propane.—(8.2).

Investigation of the errors of an inertial guidance system during satellite re-entry. J. S. White. N.A.S.A. T.N. D-322. *Aug.* 1960.—(8.2×26×25.1).

Preliminary investigation of a paraglider. F. M. Rogallo et al. *N.A.S.A. T.N. D*-443. *Aug.* 1960.—(8.2×25×3)

Modulated entry. F. C. Grant. N.A.S.A. T.N. D-452. *Aug.* 1960.
The technique of modulation, or variable coefficients, is discussed and the analytical formulation is reviewed. Representative numerical results of the use of modulation are shown for the lifting and non-lifting cases. These results include the effects of modulation on peak acceleration, entry corridor, and heat absorption. Results are given for entry at satellite speed and escape speed.—(8.2×1.8.0.1×1.9.1×25.1).

AVIATION MEDICINE

See EXTRA-ATMOSPHERIC TECHNOLOGY

ELECTRONICS

See AERODYNAMICS—TESTING AND INSTRUMENTS
POWER PLANTS

FLIGHT TESTING

See AERODYNAMICS—STABILITY AND CONTROL

FUELS AND LUBRICANTS

See also THERMODYNAMICS

Use of less reactive materials and more stable gases to reduce corrosive wear when lubricating with halogenated gases. D. H. Buckley and R. L. Johnson. N.A.S.A. T.N. D-302. *Aug.* 1960.
The gases CF_2Cl-CF_2Cl, CF_2Cl_2, and CF_2Br-CF_2Br were used to lubricate metals, cermets and ceramics at temperatures to 1400°F.—(14.3×23.1).

HYDRODYNAMICS

See AERODYNAMICS—WINGS AND AEROFOILS

Experimental investigation of two low-drag supercavitating hydrofoils at speeds up to 200 feet per second. K. W. Christopher and V. E. Johnson. N.A.S.A. T.N. D-436. *Aug.* 1960.
Two supercavitating hydrofoils—of aspect ratios one and three—designed for low drag characteristics have been tested at or near zero cavitation numbers. The tests were made over a range of depths of submersion from 0 to approximately one chord and of angles of attack from 4° to 20°.—(17.1×17.2).

MATERIALS

See also POWER PLANTS
STRUCTURES—THEORY AND ANALYSIS

A parameter to represent the mechanical properties of aluminium alloys after soaking at elevated temperatures. W. A. P. Fisher. C.P. 506. 1960.
Test data on the tensile mechanical properties of two structural aluminium alloys after heating for long periods at fixed temperatures are correlated on the assumption that the degree of over-ageing can be expressed by the value of the 0·1 per cent proof stress at room temperature after heating.—(21.2.2).

A phenomenological theory for the transient creep of metals at elevated temperatures. E. Z. Stowell. N.A.S.A. T.R. R-44. 1959.
In this theory, a metal consisting of two phases, each with its own elasticity and viscosity, will exhibit transient creep after application of a constant stress. A comparison of the transient creep curves with experimental data on four metals shows that the family of creep curves for one metal is given by using a set of constants appropriate to that metal.—(21.2).

Temperature histories in ceramic-insulated heat-sink nozzle. C. C. Ciepluch. N.A.S.A. T.N. D-300. *July* 1960.
Temperature histories were calculated for a composite nozzle wall using a simplified numerical integration calculation procedure.—(21.3.1×27.3×34.3.3).

The hot-pressing of hafnium carbide (melting point, 7030°F). W. A. Sanders and S. J. Grisaffe. N.A.S.A. T.N. D-303. *Aug.* 1960.
Effects of some hot-pressing variables (temperature, pressure, and time) on the density and grain size of ¼ in. thick hafnium carbide discs are presented.—(21.2.2).

An exploratory investigation of some factors influencing the room-temperature ductility of tungsten. J. R. Stephens. N.A.S.A. T.N. D-304. *Aug.* 1960.
Effects of the following surface treatments on the room-temperature ductility of wrought, sintered tungsten were investigated: electropolishing, surface grinding, and abrading the electropolished surface with emery paper. The effect of deflection rate on the bend ductility of electropolished and as-received specimens was also evaluated.—(21.2×21.5).

Effects of various aging heat treatments and solution-annealing and aging heat treatments on tensile, creep, and stress-rupture strengths of Inconel X sheet to temperatures of 1400°*F. F. W. Schmidt* et al. *N.A.S.A. T.N. D*-374. *June* 1960.
The effects of six different heat treatments are presented on the tensile strength, on the minimum creep rate, on the 0·2 per cent creep strength in 5 and 10 hours, and on the 10- and 100-hour stress-rupture strength of Inconel X sheet at 1,000°, 1,200°, and 1,400°F.—(21.2.1×21.5).

An evaluation of some current practices for short-time elevated-temperature tensile tests of metals. C. R. Manning and G. J. Heimerl. N.A.S.A. T.N. D-420. Sept. 1960.
An evaluation was made of the effect of the testing practice on the short-time elevated-temperature tensile properties of 2024-T3 aluminium-alloy, HM21A-T8 and HK31A-H24 magnesium-alloy, and 12 MoV stainless steel sheet. Constant strain rate, constant head speed, and dual strain rate tests were made.—(21.2.1×21.2.2×21.6.1).

Effect of thickness on the mechanical properties of glass-fabric-base plastic laminates. R. L. Young. F.P.L. Report 1873. *May* 1960.
The results of tension, compression, and flexure tests of eleven laminates fabricated from Scotchply 1002 epoxy resin and reinforced with 181-Volan A glass fabric, and 8 laminates fabricated from D.C. 2106 silicone resin and reinforced with 181-112 glass fabric are presented—(21.3.6×33.2.4.6).

Some properties of chromium and chromium-tungsten alloys. Part I—Compression creep tests. C. S. Landau et al. *A.R.L. Report Met.* 37. *June* 1960.
Compression creep curves have been determined in the range 750° to 950°C on pure chromium in the wrought condition, and at 900°-1,000°C on chromium-tungsten alloys, mainly in the cast condition.—(21.2.1).

Some properties of chromium and chromium-tungsten alloys. Part II—Fabrication and ductility. S. T. M. Johnstone et al. *A.R.L. Report Met* 38. *June* 1960.
Arc-melted ingots of chromium-tungsten alloys containing up to 10 per cent tungsten have been fabricated to strip by extrusion and rolling.—(21.2.1).

Some experiments on the effect of time at temperature on the room temperature reversed bending fatigue characteristics and on the tensile strength of 24S-T Alclad aluminum alloy. J. A. Dunsby. N.R.C. Report M.S. 102. *Aug.* 1960.
Experiments are described in which specimens of 24 S-T aluminium alloy were held at temperatures of 400°F or 300°F for periods ranging from 1¼ to 100 hours before conducting room temperature reversed bending fatigue or tensile tests.—(21.2.2×31.2.2.3.1.8).

MECHANICAL ENGINEERING

See FUELS AND LUBRICANTS

MISSILES

See also AERODYNAMICS—BOUNDARY LAYER
STABILITY AND CONTROL
THERMO-AERODYNAMICS
AIRCRAFT OPERATION
EXTRA-ATMOSPHERIC TECHNOLOGY

Analytical investigation of the dynamic behaviour of a non-lifting manned reentry vehicle. J. H. Lichtenstein. N.A.S.A. T.N. D-416. Sept. 1960.
This investigation encompassed the effects of aerodynamic derivatives and spin rate, re-entry angle and velocity, geostrophic winds, and a drogue-parachute stabiliser on the vehicle as it descended through the atmosphere.—(25.1×1.8.0.1).

Investigation of the flow over simple bodies at Mach numbers of the order of 20. *A. Henderson. N.A.S.A. T.N. D-449. Aug.* 1960.—(25.2×1.6.1×1.2.3.1×8.2).

NAVIGATION

See also EXTRA-ATMOSPHERIC TECHNOLOGY

Problems involved in an emergency method of guiding a gliding vehicle from high altitudes to a high key position. J. W. Jewel and J. B. Whitten. N.A.S.A. T.N. D-438. Aug. 1960.
An investigation has been conducted to determine the problems involved in an emergency method of guiding a gliding vehicle from high altitudes to a high key position (initial position) above a landing field. Starting test altitudes varied from 30,000 to 46,500 ft., and starting positions ranged 8·4 to 67 nautical miles from the high key. Specified altitudes of the high key were 12,000, 10,000, or 4,000 ft.—(26×5×8.2).

POWER PLANTS

See also AERODYNAMICS—FLUID DYNAMICS
INTERNAL FLOW
AIRCRAFT OPERATION
MATERIALS
THERMODYNAMICS

Experimental evaluation of rocket exhaust diffusers for altitude simulation. J. N. Sivo et al. *N.A.S.A. T.N. D-298. July* 1960.
An experimental investigation of rocket exhaust diffusers has been conducted to evaluate various methods of minimising the overall pressure ratio (from chamber to ambient pressure) required to establish and maintain full expansion of the nozzle flow (altitude simulation). Exhaust-diffuser configurations investigated were cylindrical diffusers, diffusers with contraction, and diffusers including a right-angle turn. Primary nozzles of various area ratios and types, including two clustered configurations, were used. Air was the working fluid, except for two check points obtained with JP-4 fuel and liquid-oxygen rocket engines and cylindrical diffusers.—(27.3.1).

Spatial characteristics of water spray formed by two impinging jets at several jet velocities in quiescent air. H. H. Foster and M. F. Heidmann. N.A.S.A. T.N. D-301. July 1960.
Sprays formed by two 0·089 in. impinging water jets in quiescent air were studied for a range of jet velocities typical of those used in current rocket engine design practices. A survey around the point of jet impingement was made with photographic techniques.—(27.3×1.4).

Static thrust of an annular nozzle with a concave central base. B. W. Corson and C. E. Mercer. N.A.S.A. T.N. D-418. Sept. 1960.—(27.3.1).

A preliminary investigation on the destruction of solid-propellant rocket motors by impact from small particles. D. J. Carter. N.A.S.A. T.N. D-422. Sept. 1960.
An investigation was conducted to determine the effects of impacting spherical projectiles on simulated solid-propellant rocket motors over a range of ambient pressures from sea level to 0·12 in. of mercury (approximately 125,000 ft.). Spheres from 1/16 to 7/32 in. diameter were fired into simulated rocket motors containing T-22 propellant. Simulated case thicknesses varied from 1/30 to 1/8 in.—(27.3×8.2×21.2).

A graphical method for estimating ion-rocket performance. T. W. Reynolds and J. H. Childs. N.A.S.A. T.N. D-466. Aug. 1960.
Equations relating the critical temperature and ion current density for surface ionisation of cesium on tungsten are derived for the cases of zero and finite electric fields at the ion-emitting surface. These equations are used to obtain a series of graphs that can be used to solve many problems relating to ion-rocket theoretical performance.—(27.2×11×8.2).

Some tests of a resonant combustion chamber. D. G. Stewart and C. N. King. A.R.L. Report M.E. 94. *July* 1960.
Oscillatory combustion of the type existing in gas fired "tunnel" burners has been achieved in a simple metallic tube using kerosine as fuel. Investigations have been made into the effects of tube length, throat diameter, throat shape and the use of ceramic tubing.—(27.4×34.1.1).

REFERENCE LITERATURE

See STRUCTURES—THEORY AND ANALYSIS

FATIGUE

See MATERIALS
STRUCTURES—THEORY AND ANALYSIS

SCIENCE—GENERAL

See also AERODYNAMICS—COMPRESSIBLE FLOW
EXTRA-ATMOSPHERIC TECHNOLOGY
THERMODYNAMICS

On influence functions in the theory of forced vibrations of membranes. I. Torbe and D. I. G. Jones. U.S.A.A. Report 115. *July* 1960.—(32.2.3×33.1.2).

Natural convection of an electrically conducting fluid in the presence of a magnetic field. P. S. Lykoudis. School Aero. Eng. Purdue Univ. Report A-60-1. Aug. 1960.
The case of a vertical hot plate surrounded by an electrically conducting fluid is examined in the presence of a magnetic field acting in a direction perpendicular to the induced movement caused by the buoyant forces.—(32.2.1).

STRUCTURES

LOADS *see also* AEROELASTICITY
AIRCRAFT OPERATION
SCIENCE—GENERAL

Flutter research on skin panels. E. E. Kordes et al. *N.A.S.A. T.N. D-451. Sept.* 1960.
Representative experimental results are presented to show the current status of the panel flutter problem. Results are presented for unstiffened rectangular panels and for rectangular panels stiffened by corrugated backing. Flutter boundaries are established for all types of panels when considered on the basis of equivalent isotropic plates. The effects of Mach number, differential pressure, and aerodynamic heating on panel flutter are discussed. A flutter analysis of orthotropic panels is presented.—(33.1.2×33.2.4.5×33.2.4.6×2).

Interface damping at riveted joints (Part I—Theoretical analysis). D. J. Mead and D. C. G. Eaton. U.S.A.A. Report 153. *Aug.* 1960.
A theoretical examination is made of the energy dissipated per cycle in a riveted lap joint having a visco-elastic interfacial layer and being subjected to longitudinal harmonic loading. Certain simplifying assumptions have been made. The magnitude of this energy dissipation has been computed for a wide range of joint dimensions, dynamic properties of the layer, and rivet stiffnesses. Maximised values of the energy dissipation are obtained together with the optimum joint configurations. Simple design rules have been formulated for maximum interface damping.—(33.1.2×33.2.4.13.10).

THEORY AND ANALYSIS *see also* AERODYNAMICS—INTERNAL FLOW
MATERIALS
STRUCTURES—LOADS

Supersonic flutter of cylindrical shells. D. J. Johns. C.o.A. Note 104. *July* 1960.
The general theories of thin elastic cylindrical shells as derived by Love and Novozhilov are compared and then used in a simple binary flutter analysis which permits the existence of both axial and circumferential waves of deformation. Comparisons are made with other published results.—(33.2.4×2).

Stress distribution in pressurized cabins: an experimental study by means of Xylonite models. T. H. Richards. C.P. 503. 1960.
The effect of openings on the stresses in plain, pressurised circular cylindrical shells is discussed. A review of relevant theory is presented and the design, construction and testing of Xylonite models is described. Results are discussed, and conclusions drawn, with proposals for future work.—(33.2.3.3×33.2.4.3.7).

Investigation of the buckling strength of corrugated webs in shear. J. P. Peterson and M. F. Card. N.A.S.A. T.N. D-424. June 1960.
Design charts are presented from which the buckling strength of corrugated shear webs can be determined. The charts are applicable to webs with supported edges in which the edge rotations of the web along lines of support may range from unrestrained (simply supported edges) to completely restrained (clamped edges). The results of shear tests on seven beams with corrugated webs are discussed.—(33.2.4.6.4×33.4.1).

List of Publications on Mechanical Properties of Structural Uses of Wood and Wood Products. F.P.L. Report 200. *Jan.* 1960.—(33.2.4.5×21.2.3×21.3.6×30.1).

Torsion of sandwich panels of trapezoidal, triangular, and rectangular cross sections. S. Cheng. F.P.L. Report 1874. *June* 1960.
A theoretical analysis is made of the torsion of a sandwich panel with a trapezoidal cross section. The facings are of uniform and equal thickness. The core is tapered so that the cross section taken perpendicular to the x axis is a trapezoid and that perpendicular to the y axis is rectangular.—(33.2.4.6.5).

Application of the matrix force stress analysis to a stress diffusion problem on the Mustang wing. L. J. Schmid. A.R.L. Note S.M. 265. *Feb.* 1960.
In assessing the fatigue characteristics of a Mustang wing a knowledge of the stresses in several critical fatigue failure areas under 10 per cent ultimate static load is required. Several wings were tested under the required static loads and strains were measured in the appropriate areas. Stresses in the critical fatigue failure areas were also estimated by linear interpolation of published stress results. The results of a stress diffusion calculation by the matrix force method are published.—(33.2.3.1×31.2.4.2×3.6).

Small deflection theory of flat plates using complex variables. Part III—Plates with specified edge forces and moments and simply supported plates. P. D. Jones. A.R.L. Report S.M. 266. *May* 1959.
Boundary equations for plates with specified edge forces and moments and simply supported plates are formulated and general methods of solution for arbitrary loading conditions are discussed. Examples illustrate how these general methods may be applied to specific plate problems.—(33.2.4.5).

Membrane theory of large spherical radomes. A. L. M. Grzedzielski. N.R.C. Report LR-278. May 1960.
Two methods of stress analysis of large spherical radomes are developed: an elementary desk calculator analysis, based on the restrictive assumption of airload being axially symmetric with respect to the wind direction, and a general unit load method, making use of a medium size digital computer. Both account for an elastic support at the foundation.—(33.2.4.11).

WEIGHT ANALYSIS AND CONTROL *see* THEORY AND ANALYSIS

THERMODYNAMICS

See also AERODYNAMICS—BOUNDARY LAYER
INTERNAL FLOW
MATERIALS
POWER PLANTS

A study of the combustion of aluminum borohydride in a small supersonic wind tunnel. H. Allen and E. A. Fletcher. N.A.S.A. T.N. D-296. July 1960.—(34.1.1×32.1×14).

Real-gas correction factors for hypersonic flow parameters in helium. W. D. Erickson. N.A.S.A. T.N. D-462. Sept. 1960.
The hypersonic flow parameters for helium have been calculated from the thermodynamic properties of helium for stagnation temperatures from 0°F to 600°F and stagnation pressures up to 6,000 lb./in.² abs. The results are presented as simple correction factors which must be applied to the ideal-gas parameters.—(34.1×1.2.3.1×8.2×32.2.2×1.12.1.3).

Liquid-phase heat-release rates of the systems hydrazine-nitric acid and unsymmetrical dimethylhydrazine-nitric acid. D. Somogyi and C. E. Feiler. N.A.S.A. T.N. D-469. Sept. 1960.
The initial rates of heat release were determined in a bomb calorimeter under conditions of forced mixing. The mixing was varied by varying injection velocities. Fuel-oxidant weight ratio was also varied. A possible combustion mechanism based on reaction at interfaces is discussed.—(34.1.1×27.3×14.3).

Prandtl number, thermal conductivity, and viscosity of air-helium mixtures. E. R. G. Eckert et al. *N.A.S.A. T.N. D-533. Sept.* 1960.
Prandtl number measurements of air-helium mixtures were performed at a temperature of 270°K for mixtures ranging from pure air to pure helium. Various methods of predicting the thermal conductivity and viscosity of binary mixtures were investigated and results are compared with the experimental results. Dimensionless conductivities, viscosities, and Prandtl numbers were calculated from 200° to 1,500°K for mixtures ranging from pure air to pure helium.—(34.1×32.2×32.1).